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FINAL REPORT
SYSTEM DESIGN OF THE PIONEER VENUS SPACECRAFT

VOLUME 8
COMMAND/DATA HANDLING SUBSYSTEMS STUDIES

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PREFACE

The Hughes Aircraft Company Pioneer Venus final report is based on study task reports prepared during performance of the "System Design Study of the Pioneer Spacecraft." These task reports were forwarded to Ames Research Center as they were completed during the nine months study phase. The significant results from these task reports, along with study results developed after task report publication dates, are reviewed in this final report to provide complete study documentation. Wherever appropriate, the task reports are cited by referencing a task number and Hughes report reference number. The task reports can be made available to the reader specifically interested in the details omitted in the final report for the sake of brevity.

This Pioneer Venus Study final report describes the following baseline configurations:

- "Thor/Delta Spacecraft Baseline" is the baseline presented at the midterm review on 26 February 1973.
- "Atlas/Centaur Spacecraft Baseline" is the baseline resulting from studies conducted since the midterm, but prior to receipt of the NASA execution phase RFP, and subsequent to decisions to launch both the multiprobe and orbiter missions in 1978 and use the Atlas/Centaur launch vehicle.
- "Atlas/Centaur Spacecraft Midterm Baseline" is the baseline presented at the 26 February 1973 review and is only used in the launch vehicle utilization trade study.

The use of the International System of Units (SI) followed by other units in parentheses implies that the principal measurements or calculations were made in units other than SI. The use of SI units alone implies that the principal measurements or calculations were made in SI units. All conversion factors were obtained or derived from NASA SP-7012 (1969).

The Hughes Aircraft Company final report consists of the following documents:

<u>Volume 1 - Executive Summary</u> - provides a summary of the major issues and decisions reached during the course of the study. A brief description of the Pioneer Venus Atlas/Centaur baseline spacecraft and probes is also presented.

Volume 2 - Science - reviews science requirements, documents the science-peculiar trade studies and describes the Hughes approach for science implementation.

Volume 3 - Systems Analysis - documents the mission, systems, operations, ground systems, and reliability analysis conducted on the Thor/Delta baseline design.

Volume 4 - Probe Bus and Orbiter Spacecraft Vehicle Studies - presents the configuration, structure, thermal control and cabling studies for the probe bus and orbiter. Thor/Delta and Atlas/Centaur baseline descriptions are also presented.

Volume 5 - Probe Vehicle Studies - presents configuration, aerodynamic and structure studies for the large and small probes pressure vessel modules and deceleration modules. Pressure vessel module thermal control and science integration are discussed. Deceleration module heat shield, parachute and separation/despin are presented. Thor/Delta and Atlas/Centaur baseline descriptions are provided.

Volume 6 - Power Subsystem Studies

Volume 7 - Communication Subsystem Studies

Volume 8 - Command/Data Handling Subsystems Studies

Volume 9 - Altitude Control/Mechanisms Subsystem Studies

Volume 10 - Propulsion/Orbit Insertion Subsystem Studies

Volumes 6 through 10 - discuss the respective subsystems for the probe bus, probes, and orbiter. Each volume presents the subsystem requirements, trade and design studies, Thor/Delta baseline descriptions, and Atlas/Centaur baseline descriptions.

Volume 11 - Launch Vehicle Utilization - provides the comparison between the Pioneer Venus spacecraft system for the two launch vehicles, Thor/Delta and Atlas/Centaur. Cost analysis data is presented also.

Volume 12 - International Cooperation - documents Hughes suggested alternatives to implement a cooperative effort with ESRO for the orbiter mission. Recommendations were formulated prior to the deletion of international cooperation.

Volume 13 - Preliminary Development Plans - provides the development and program management plans.

Volume 14 - Test Planning Trades - documents studies conducted to determine the desirable testing approach for the Thor/Delta spacecraft system. Final Atlas/Centaur test plans are presented in Volume 13.

Volume 15 - Hughes IR&D Documentation - provides Hughes internal documents generated on independent research and development money which relates to some aspects of the Pioneer Venus program. These documents are referenced within the final report and are provided for ready access by the reader.

Data Book - presents the latest Atlas/Centaur Baseline design in an informal tabular and sketch format. The informal approach is used to provide the customer with the most current design with the final report.

CONTENTS

1. SUMMARY

Study tasks for the command and data handling subsystems have been directed to:

1) Determining ground data systems (GDS) interfaces and deep space network (DSN) changes, if required

2) Defining subsystem requirements

3) Surveying existing hardware that could be used or modified to meet subsystem requirements

4) Establishing a baseline design

Study of the existing GDS led to the conclusion that the Viking configuration GDS can be used with only minor changes required for the Pioneer Venus baseline. Those changes required are associated with providing a predetection recording capability used during probe entry and descent.

Subsystem requirements were first formulated with sufficient latitude so that surveys of existing hardware could lead to low cost hardware which, in turn, could modify more narrowly defined subsystem requirements.

To a large extent, existing hardware has been found that will meet performance requirements for the spacecraft subsystems. The hardware selected is largely from the OSO-I program, which provides equipment specifically developed for interfacing various scientific experiments for a space mission. Hardware for the large and small probe vehicles is mostly of new design, due to stringent weight, power, and volume limitations. There is a great deal of commonality between the large and small probes, and at the module level, many circuits come from the OSO-I program.

Originally (as of the midterm presentation), there was little difference in the command and data handling hardware developed for the Thor/Delta or Atlas/Centaur versions of the program. Functional requirements and performance were identical in all respects. Boost vehicle payload limitations of Thor/Delta forced a different product design that made extensive use of LSI for new circuit designs. In addition, the large and small probe data rates and formats were different for the two versions due to different descent velocities. These were the extent of the differences.

The program decision to utilize the Atlas/Centaur launch vehicle rather than the Thor/Delta, and the coincident decisions to utilize the same launch opportunity for both the multiprobe and orbiter missions and change the science payload, necessitated some baseline changes in the mission and hardware designs which are summarized as follows:

1) Two spacecraft in transit to Venus at same time - requires some means of assuring that the spacecraft can be separately addressed and commanded. Also requires separation of downlink frequencies and makes desirable a spacecraft identifier within the data stream.

2) New science payload - due to the great increase in science data rates at periapsis, some changes in science data formats, spacecraft clock rates, and data storage accommodation have been required.

The majority of the material and trade studies discussed in this report (Sections 4 and 5) were generated prior to the program changes discussed above. In general, none of the trade studies were affected by launch vehicle selection considerations up until these program changes occurred; therefore, no general attempt has been made to update the trade studies with the affects described. The material contained in Section 6, however, does represent updated current baseline designs for the Atlas/Centaur 1978 missions.

Tables 1-1 to 1-3 tabulate the current Atlas/Centaur baseline characteristics and hardware source.

The spacecraft command subsystem will handle over 192 pulse commands and 12 magnitude commands. A 16 bit magnitude command is utilized. The command memory can store 85 24- bit words. Each memory unit can represent pulse, magnitude, or time delay information. The command word length is 36 bits, which can accommodate either a real time magnitude command or a word of command memory data.

The probe vehicles operate from commands stored in fixed sequences. Sequences of commands are initiated from a sequencer both as the result of the prestored timing sequence and/or the result of selected descent events (acceleration and pressure). The stored sequences include a short (15 minute) systems test to be used subsequent to separation from the launch vehicle; a long dormant coast phase (up to 21 days); preentry turn on; and descent sequence control. Pulse type commands, only, are initiated from the sequence.

Data handling is required to acquire, encode, format, store, modulate, and deliver data to the telecommunications subsystem for subsequent transmission to earth. Each spacecraft and probe vehicle produces a single composite telemetry stream for transmission to the earth. Prior to separation from the probe bus, both the large and small probes provide composite

TABLE 1-1. SPACECRAFT MODULAR COMMAND SUBSYSTEM HARDWARE DERIVATION AND CHARACTERISTICS

Unit (Quantity)	Hardware Derivation		Hardware Characteristics		
	Atlas/Centaur Probe Bus	Atlas/Centaur Orbiter	Mass (Weight), kg (lb)	Power, W	Volume, cm^3 (in^3)
Command demodulator (2)	Motorola (Uses 98 percent Viking orbiter circuits; MSI)	Same	2.5 (5.6)	3.0	3,080 (188)
Central decoder (2)	New design (Uses 50 percent OSO circuits; MSI)	Same	4.3 (9.4)	3.6	6,000 (366)
Command output module (6) (remote decoder)	OSO (six single units; existing LSI)	Same	1.8 (4.2)	0.3	1,720 (105)
Pyro control unit (2)	New design (Uses 70 percent existing circuits)	Same	1.2 (2.5)	-0-	4,260 (260)
		Probe bus totals	9.8 (21.7)	6.9	15,060 (919)
		Orbiter totals	Same	Same	Same

TABLE 1-2. SPACECRAFT MODULAR DATA HANDLING SUBSYSTEM HARDWARE DERIVATION AND CHARACTERISTICS

Unit (Quantity)	Hardware Derivation		Hardware Characteristics*		
	Atlas/Centaur Probe Bus	Atlas/Centaur Orbiter	Mass (Weight), kg (lb)	Power, W	Volume cm^3 (in^3)
Data Input Module (3) (remote multiplexer)	OSO (three dual units; existing LSI)	Same	1.1 (2.4)	0.4	900 (55)
PCM encoder (2)	OSO (two single units; existing MSI)	Same	3.1 (6.8)	2.6	3,510 (214)
Telemetry processor (2)	New design (Uses 70 percent OSO circuits; MSI)	Same	3.6 (8.0)	5.7	6,770 (413)
Data storage unit (2)	Not required	Core memory-POV (Modified electronic memories design)	--- --- 9.1 (20.0)	--- 3.0	--- --- 9,630 (588)
		Probe bus totals	7.8 (17.2)	8.7	11,180 (682)
		Orbiter totals	16.9 (37.2)	11.7	20,810 (1270)

*Where two sets of values are given, the bottom line is for the orbiter spacecraft.

TABLE 1-3. PROBE COMMAND AND DATA HANDLING SUBSYSTEM HARDWARE DERIVATION AND CHARACTERISTICS

Unit	Atlas/Centaur Hardware Derivation	Hardware Characteristics: Mass (Weight), kg (lb)	Hardware Characteristics: Power Cruise/Descent W	Hardware Characteristics: Volume cm^3 (in^3)
Large probe				
Command/data unit	New design, existing technology; MICAM construction	1.95 (4.3)	0.04/8.3	2,720 (166)
Pyro control unit	New design, existing technology; 50 percent probe bus circuits	1.18 (2.6)	(Transient only)	1,330 (81)
Acceleration switches (2)	(1) Existing design; space qualified (1) Existing design; non-space qualified	0.23 (0.5)	---	66 (4.0)
Pressure switches (2)	Existing design, high reliability; non-space qualified	0.18 (0.4)	---	25 (1.5)
Total		3.54 (7.8)	0.04/8.3	4,140 (253)
Small probe				
Command/data unit	New circuit design; 70 percent large probe circuits New product design; MICAM construction	1.54 (3.4)	0.04/4.6	2,280 (139)
Regulator unit	New package, existing circuit design	0.18 (0.4)	0.0/1.2	197 (12)
Pyro control unit	New circuit design; 80 percent large probe circuits New product design	0.55 (1.2)	(Transient only)	361 (22)
Acceleration switches (2)	(1) Same as large probe; space qualified (1) Existing design; non-space qualified	0.23 (0.5)	---	66 (4.0)
Pressure switches (2)	(1) Same as large probe (1) Existing design, high reliability; non-space qualified	0.18 (0.4)	---	25 (1.5)
Total		2.68 (5.9)	0.04/5.8	2,930 (179)

telemetry streams which can be summed and transmitted to Earth via the probe bus telecommunications subsystem.

Data is telemetered as 8 bit words. The words are grouped into minor frames of 32 words and major frames of 16 minor frames. The configuration of any minor frame (data points to be sampled) is determined by a read-only memory in the telemetry processor. For the probe bus, there are nine different minor frames, for the orbiter there are 12. Selection and order of the 16 minor frames within a major frame is determined by ground command.

The spacecraft telemetry processor contains 16 different major frames; any one of which may be selected by ground or stored command. The identification of the format selected is telemetered as part of each frame. Data rates are in multiples of 2^n between 8 and 2048 bps, depending on the mission phase and rf link margin. Capability is provided to store one telemetry format and transmit another to ground in real time.

Data for the probes is telemetered as 10-bit words. In order to maximize the science data return within the available data bandwidth, engineering and science housekeeping data subcommutators are provided. There are two data formats for each probe type.

Data rates for the large probe are 160 bps which is switched to 80 bps at 20 km from the surface. Rates of 60 bps, 30 bps, and 10 bps are implemented for the small probe. These are switched during the descent sequence to conform to the rf link transmission capability.

A data storage capability is provided on the orbiter spacecraft to gather data at rates exceeding the prevailing telecommunications capability and during planetary occultations when data transmissions are interrupted.

2. INTRODUCTION

The command and data handling subsystems' basic functions are commanding and receiving data from the science instruments, and general operation of the orbiter and multiprobe missions. Low cost configurations that meet subsystem requirements are a design goal.

The subsystems must accommodate science instrument command and data requirements. Flexibility must be incorporated into the design because of possible changes in the science instruments. Minimum changes in the DSN and other elements of the GDS are desirable; therefore, the subsystems must have a high degree of compatibility. The complete Pioneer Venus system should be easy to operate; thus, complexity must be minimized.

The central design of the subsystems should be based on the use of existing, proven hardware. This will minimize costs. Where off-the-shelf hardware cannot be used or modified to meet requirements, new designs should be directed to simple elements and simple logic to provide a low cost implementation.

An outline of the general studies performed for the command and data handling subsystems is presented in Table 2-1. A summary of these studies follows in subsequent sections of this volume. Requirements are summarized in the following section. The next section is devoted to the major subsystem trade studies which have been performed.

Command subsystem studies include command interface, execution and storage, and probe cruise timer. Data handling subsystems studies include interface, probe and orbiter data storage and probe stored data playback techniques, multiple data formats and multiple data rates.

Available subsystem hardware is then described. Subsequent sections contain discussions of the receiver selection technique, and various aspects of the pyrotechnics and their control. Finally, the results and conclusions of a study investigating use of a programmable on-board data processor are presented.

Command and data handling subsystems baseline designs have been established for both Thor/Delta and Atlas/Centaur boosted missions. Details of these baselines can be found in Sections 5 and 6 of this volume. The current subsystem baseline is described in Section 6. Section 5 represents the midterm Thor/Delta baseline, which has not been updated due to the program decision to utilize the Atlas/Centaur launch vehicle.

TABLE 2-1. STUDY OUTLINE

Task No.	Task	Hughes Report Ref. Number
CC	Command/Control	
CC1	Preliminary command list preparation	HS-507-0022-73
CC2	Analyze command storage requirements	Final Rpt
CC3	Command modulation techniques and message formats	Final Rpt
CC6	Command interfaces with experiments	HS-507-0022-148
CC7	Probe stored sequence requirements	Final Rpt
CC8	Analyze prevention of inadvertent irreversible command execution	HS-507-0022-68
CC9	Command subsystem functional design	Final Rpt
DH	Data Handling	
DH1	Data storage requirements	Final Rpt
DH2	Use of central programmable processor	Final Rpt
DH3	Multiplexer and analog/digital converter requirements	Final Rpt
DH4	Evaluate digital interface designs	HS-507-0022-149
DH5	Probe data storage	HS-507-0022-38
DH6	Probe data rate	HS-507-0022-37
DH7	Probe data handling frame optimization	Final Report
DH8	Data handling list preparation	Final Report
DH9	Bus data handling format requirements	Final Report
DH12	Data handling subsystem functional design	Final Report

3. SUBSYSTEM REQUIREMENTS

Requirements for command and data handling subsystem performance have been established by considerations of compatibility with the Deep Space Network (DSN), commanding of and receiving data from the science instruments, and operability of multiprobe and orbiter missions. The following subsections summarize the command and data handling subsystem performance requirements. This summary is based upon the Thor/Delta mission and baseline performance as defined in the midterm presentation. Updated or changed requirements based upon the current Atlas/Centaur configuration are available in Subsection 6.1.

The requirements presented here are the end product of successive iterations performed in the selection of the baseline configurations. Preliminary design on the basis of general trade studies and hardware surveys then resulted in a system outline, with more specific requirements. These requirements, in turn, were used in detailed trade studies leading to the system design. The requirements summarized in this section are generally those utilized for the trade studies and for the midterm baseline.

3.1 PROBE BUS AND ORBITER COMMAND REQUIREMENTS

The command subsystem shall demodulate, decode, and distribute ground commands to enable control of the spacecraft. It shall store commands and associated time delays for later execution in order to implement spacecraft control during periods when real time control is not possible (e.g., before initial acquisition and during orbiter occultation). Modulation shall be PCM/PSK/PM or PCM/FSK/PM for DSN compatibility. Command bit rates shall be 1, 2, 4, or 8 bps to minimize command times at varying mission communications distances and remain within the DSN allowable rates of 1 to 8 bps. The command word length shall be 36 bits, which is compatible with the DSN requirement of ≤ 72 bits and satisfies addressing, magnitude size, and timing requirements. Real time and stored commands shall be executed with a resolution of ± 0.125 sec. Command execution error contribution due to timer inaccuracy for stored commands shall be less than ± 0.125 sec/hr resulting in a total error of ≤ 0.375 sec after a 2 hr delay (required at orbiter orbit insertion). Maximum stored time delay shall be ≤ 4 hr.

TABLE 3-1. SPACECRAFT COMMAND ASSIGNMENTS

Subsystem	Pulse	Magnitude	Interlocked
Probe Bus:			
Command	7	2	0
Communications	18	0	0
Control and propulsion	17	1	1
Data handling	6	3	0
Power	60	0	0
Probes	12	4	4
Science	18	0	2
	138	10	7
Orbiter:			
Command	7	2	0
Communications	16	0	0
Control and propulsion	31	4	3
Data handling	9	3	0
Power	69	0	0
Science	41	1	2
	173	10	5

Commands shall be of three types:

1) Pulse-required for on/off switching

2) Magnitude-required for transferring serial data

3) Interlocked-required for added safety against inadvertent execution of critical functions (squib firing, thrusting, etc.)

Numbers of commands required are presented in Table 3-1. The serial data portion of the magnitude command shall be 16 bits as required for memory loading and attitude control data transfer.

Memory word size shall be 24 bits to accommodate command addressing, word identifier, and serial data requirements. The total number of memory words (command or time) shall be 64, per anticipated storage requirements at orbit insertion.

Overall command subsystem requirements are summarized in Table 3-2.

Command Word Format

A 22 bit command word format was considered in the initial study phases since it could have been compatible with existing Pioneer software and some existing hardware. Further study of command requirements revealed that a 22 bit word was undesirable to satisfy addressing requirements, particularly with regard to memory fill and other magnitude command requirements. DSN command word requirements dictated that the efficient command word be of length ≤ 72 bits. A 36 bit word was selected for the reasons that it satisfies spacecraft command requirements, and is largely compatible with existing OSO command hardware. It results in simplified spacecraft command subsystem operations since only one magnitude command is now required to fill a memory word or transfer enough bits to satisfy subsystem serial data requirements. The selected format is the following:

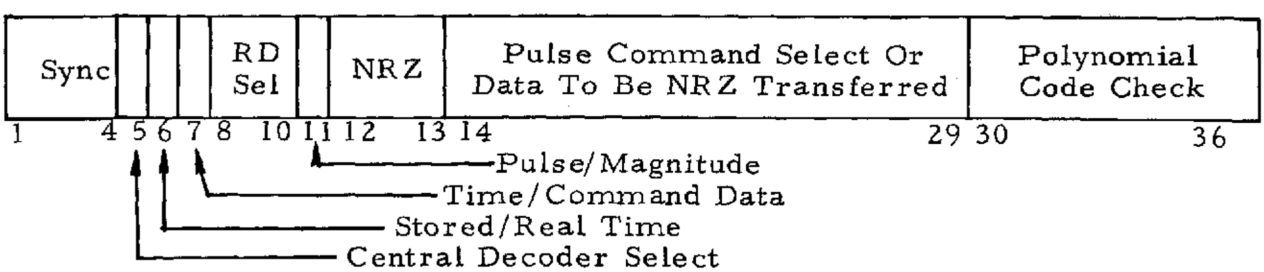


TABLE 3-2. COMMAND SUBSYSTEM REQUIREMENTS

Multiprobe bus

- Demodulate, decode, distribute ground commands
- Store commands for later execution, 17 words minimum storage
- PCM/PSK/PM or PCM/FSK/PM
- 1, 2, 4, or 8 bps command rate
- 36 bit command word
- Pulse commands:
  - 120 engineering
  - 18 experiment
- Magnitude commands:
  - 10 engineering
- Interlocked commands:
  - 7 engineering
- Threshold
  - 1×10^{-5} BER
- Probability of executing false command at threshold
  - 1×10^{-9}

Orbiter

- Demodulate, decode, distribute ground commands
- Store commands for later execution, 25 words minimum storage
- PCM/PSK/PM or PM/FSK/PM
- 1, 2, 4, or 8 bps command
- 36 bit command word
- Pulse commands:
  - 132 engineering
  - 41 instrument
- Magnitude commands:
  - 9 engineering
  - 1 instrument
- Interlocked commands:
  - 3 engineering
  - 2 instrument
- Time
  - 10 sec in 24 hour
  - 0.5 sec in 1 hr
- Threshold
  - 1×10^{-5} BER
- Probability of executing false command at threshold
  - 1×10^{-9}

Command bit assignments are the following:

Bits	Function
1 to 4	Sync
5	Central decoder select
6	Indicates whether remainder of word is to be stored or used in real time
7	Indicates whether bits 14 to 29 are command or time information if the word is going to storage (the "time" indication is not allowed if bit 6 indicates "real time")
8 to 10	Addresses 1 of 8 remote decoders (All 8 need not be used)
11	Indicates whether bits 14 to 29 address 1 of 64 pulse command lines at a remote decoder, or are to be transferred as serial data
12, 13	Addresses 1 of 4 lines from remote decoder over which serial data is to be transferred
14 to 29	Addresses 1 of 64 pulse command lines at a remote decoder, or contains 16 bits of serial data to be NRZ transferred
30 to 36	Used in the polynomial check to detect as a minimum any one or two bit errors occurring in bits 5 through 29

DSN Imposed Requirements

Table 3-3 summarizes the pertinent spacecraft DSN support requirement and the DSN capability.

The DSN command system employs either FSK or PSK modulation.

Subcarrier frequency is constrained to lie between 100 Hz and 1.0 MHz with 0.1 Hz resolution. A subcarrier frequency of 512 Hz is selected based upon existing hardware requirements.

DSN compatible data rates are 1 to 8 bps with a resolution of 1×10^{-5} bps. Data rates of 1, 2, 4, and 8 bps are compatible with mission requirements and communications capabilities, and are therefore selected as baseline. Resulting memory fill times are the following (assuming an 85 word x 24 bit memory).

Bit Rate	Fill Time
1	51 min
2	25.5 min
4	12.75 min
8	6.375 min

DSN uncertainty in the transmission of command information for PSK modulation is 1 bit time.

The maximum time required to reconfigure the DSN command subsystem is < 30 min.

An idle sequence of any repetitive modulo 24 bit pattern or subcarrier only can be generated at any DSS.

TABLE 3-3. DSN SUPPORT REQUIREMENTS – COMMAND

Function	Spacecraft Requirement	DSN Capability
• Subcarrier frequency	TBD	100 Hz to 1.0 MHz
• Type	Biphase PSK or FSK	PSK or FSK
• Rate	1, 2, 4, 8 bps	1 to 8 bps
• Word length	36 bits	< 72 bits

3.2 PROBE COMMAND REQUIREMENTS

Mission Imposed Requirements

The probe mission consists of a cruise period aboard the probe bus spacecraft for approximately 100 days. At 20 days before entry the probes are released from the spacecraft at which time they begin operations from an onboard sequencer. During the 100 day preseparation cruise phase probe test events are controlled by commands into the probe from the bus. There are five such commands for each probe:

1) Start probe test
2) Stop probe test
3) Start probe timer
4) Stop probe timer
5) Load probe timer (magnitude)

Following separation, control is from the sequencer that issues commands at a preprogrammed time, or upon sensing an event.

Timing Requirements

Timer resolution and accuracy are determined by the mission requirement to apply power to probe subsystems following the 20 day coast to within ±20 sec of the desired time. Postentry timing requirements delineated in Tables 3-4 and 3-5 are less constraining than the above requirement.

Command Generation Requirements

Science data reception and probe spacecraft control requirements dictate the large and small probe descent command sequences presented in Tables 3-4 and 3-5, respectively. Overall command generation requirements are summarized below:

	Events	Number of Commands	Total Commands Issued
Large probe	27	43	77
Small probe	7	15	25

TABLE 3-4. LARGE PROBE DESCENT COMMAND SEQUENCE

Subsequence	Time		Commands	Initiated By
Timer initiation	R+0	$E-20^d$		Bus command
Postsepara-tion test	S+0	$E-20^d$	Engineering electronics-On RF-On	Separation switch
End test	$S+10^m$ $\pm 5^s$	$Sep+10^m$	RF-Off Engineering electronics-Off	Timer delayed 10^m from separation switch
Pre-entry I	$R+18^d$ $\pm 1^h$	$E-2^d$	Engineering electronics-On Planet flux heater-On Engineering electronics-Off	Timer
Pre-entry II	$R+20^d$ $\pm 20^s$ = P+0	$E-15^m$	Engineering electronics-On RF-On Sensors-On	Timer
Pre-entry III	$P+10^m$ $\pm 1^m$	$E-5^m$	Level II set Deceleration module-On Format I select Data rate I select	Timer
Blackout I	B+0	$E+5^s$	Format II select Data rate II select Re-initiation timer	0.5 g switch
Blackout II		$E+6^s$	Arm monitor Re-initialize timer (burn)	3.0 g switch

Table 3-4 (continued)

Subsequence	Time		Commands	Initiated By
Postblackout	$B+10^s$ $\pm 1^s$	$E+15^s$	Format II select Data rate III select Deceleration module-Off	Timer
3g switch closure	C+0	$E+21^s$	Re-initialize timer	3g switch Timer switch
Chute deploy	$C+4^s$ $\pm 0.1^s$	$E+25^s$	PCU-On Fire chute deploy squibs	Timer
Interface disconnect	$C+5^s$ $\pm 0.1^s$	$E+26^s$	Fire IPD squibs	Timer
Aeroshell jettison	$C+5.5^s$ $\pm 0.1^s$	$E+26.5^s$	Fire jettison squibs	Timer
Descent	$C+6^s$ $\pm 0.1^s$	$E+27^s$	Format IV select Data rate IV select Level II set Science-On Window heater-On Fire breakoff hat PCU-Off	Timer
Chute jettison		$E+9.6^m$	PCU-On Fire chute jettison squibs PCU-Off Stepup mass spectrometer power	0.5 Atm pressure switch
Window jettison	$C+14.2^m$ $\pm 8.0^m$	$E+14.5^m$	PCU-On Fire window jettison squibs PCU-Off	Timer

Table 3-4 (continued)

Subsequence	Time		Commands	Initiated By
Data rate reduction		E+28. 2^m	Data rate II select Level III set	25 Atm. pressure switch
Impact	0+53. 7^m ±1. 0^m	E+51. 2^m	Format I select Window heater-Off Planet flux heater-Off Sequencer-Off	Timer
Mass spectrometer events			Ten of the following: PCU-On Fire inlet valve squib PCU-Off	Mass spectrometer (asynchronous command generation possibly coincident with other events)

3.3 PROBE BUS AND ORBITER DATA HANDLING REQUIREMENTS

Two constraints on data system requirements and design have been DSN compatibility and science experiment data requirements. The impact of these constraints and the trade study results are summarized in the following paragraphs.

Table 3-6 summarizes the overall spacecraft data handling requirements for the probe bus and for the orbiter.

Table 3-7 summarizes the assignment of engineering and science telemetry data channels to minor frames on the probe bus and orbiter. These assignments are required to meet engineering and science data sampling requirements.

Table 3-8 provides telemetry mode (or major frame) configurations that are required to meet data sampling requirements at critical mission phases.

DSN Compatibility

Requirements for DSN compatibility and associated spacecraft parameters are summarized in Table 3-9. The prime requirement has been

TABLE 3-5. SMALL PROBE DESCENT COMMAND SEQUENCE

Subsequence	Time		Commands	Initiated By
Timer initiation	R+0	E-20^d		Bus command
Magnetic + rf calibration	S+0	E-20^d	Engineering electronics-On RF-On Science-On Format I select	Separation switch
End test	S+10^m ±5^s	Sep+10^m	Science-Off RF-Off Engineering electronics-Off	Timer delay 10^m after separation
Pre-entry I	R+20^d ±2. 0^s	E-45^m	Engineering electronics-On	Timer
Pre-entry II	R+20^d ±2. 0^s	E-15^m	RF-On Science-On PCU-On Fire despin thrusters PCU-Off	Timer
Pre-entry III	R+20^d ±2. 0^s	E-$2^m 20^s$	Format II select RF-Off	Timer
Descent	D+0	E+19. 5^s to E+32. 5^m	RF-On Format I select Window heater-On PCU-On Fire temperature probe/nephelometer cover squib PCU-Off Re-initialize timer	2. 0 g switch
Impact	D+74^m ±1^m	E+74. 2^m to E+74. 9^m	Window heater-Off Format II science	Timer

TABLE 3-6. OVERALL SPACECRAFT DATA HANDLING REQUIREMENTS

Multiprobe bus

- Multiplex, format, encode and modulate spacecraft and experiment data
- PCM/PSK/PM
- Convolutional encoding
- 8 to 2048 bps data rates
- Data frames:
  - 9 frames; 32, 8 bit words
- Operating modes:
  - Real time
- Formats (four nominal frame combinations; variations selectable on command)
  - Cruise science (16 to 84 bps)
  - Entry science (2048 bps)
  - Thruster firing (8 to 128 bps)
  - Engineering interrogation (16 to 64 bps)
- Engineering channels
  - 113 analog
  - 13 serial digital
  - 98 discrete
- Experiment channels
  - 31 analog
  - 13 serial digital
  - 1 discrete

Orbiter

- Multiplex, format, encode, and modulate spacecraft and experiment data
- PCM/PSK/PM
- Convolutional encoding
- 8 to 2048 bps data rate
- Data frames:
  - 12 frames; 32, 8 bit words

Table 3-6 (continued)

Operating modes: Real time Stored Stored and real time Memory readout Memory readout and real time
Formats (nine nominal frame combinations; variations selectable on command) Cruise science (16 to 64 bps) Apoapsis science (2) (64 to 128 bps) Periapsis science (64 to 128 bps) Store (3) (5 to 428 bps) Thruster firing (8 to 128 bps) Engineering interrogation (16 to 64 bps)
Engineering channels 137 analog 10 serial digital 46 discrete
Experiment channels 7 analog 5 serial digital 1 discrete
361 kilobits data storage

to minimize changes to the DSN and to be compatible with the multimission capability so that data from the spacecraft can be processed in real time and transmitted over high speed data lines (HSDL) to the ARC Pioneer Mission Operations Center (PMOC). Except for the requirement to provide predetection recording capability for probe entry and descent (discussed in Volume 3), no changes are required in the DSN.

Subcarrier frequency selection was accomplished on the basis of DSN and bit rate compatibility and requirements of available transmitter hardware. As noted in Table 3-9 the DSN compatibility is basically between 20 and 45 kHz. The spacecraft subcarrier is selected at 16 times this maximum bit rate of 2,048 bps. Probe subcarrier frequencies are selected above 20 kHz (the minimum capability of the selected, available hardware transmitter) and at frequencies of integer multiples of the bit rates. Modulation type PCM/PSK/PM is dictated by the DSN MMC.

TABLE 3-7. MINOR FRAME DATA CHANNEL ASSIGNMENTS

Common to Probe Bus and Orbiter Spacecraft

Engineering A, B, C, D

2W	1W	29W
SYNC	FRAME ID	ENGINEERING HOUSEKEEPING

(Frame Words)

TCM A, B

2W	1W	29W
SYNC	FRAME ID	ENGINEERING HOUSEKEEPING

(Frame Words)

CMR (Command Memory Recycle)

2W	1W	14W	14W	1W
SYNC	FRAME ID	MEMORY A	MEMORY B	SPARE

(Frame Words)

Orbiter Spacecraft

SCI A

2W	1W	7W	7W	8W	7W
SYNC	FRAME ID #	MAGN	SOLAR WIND	UV SPECT	SPARE

(Frame Words)

Table 3-7 (continued)

Orbiter (Continued)

SCI B (Housekeeping)

2W	1W	1W	1W	1W	1W	1W	1W	1W	1W	1W	
SYNC	FRAME ID #	MH1	MH2	MH3	MH4	LPH1	LPH2	LPH3	NMSH1	NMSH2	(Frame Words)

1W	1W	1W	1W	1W	1W	1W	1W	1W	
NMSH3	NMSH4	IMSH1	IMSH2	IMSH3	UVSH1	UVSH2	UVSH3	UVSH4	(Frame Words)

1W	1W	1W	1W	1W	1W	1W	1W	1W	
UVSH5	UVSH6	IRRH1	IRRH2	IRRH3	IRRH4	SWH1	SWH2	SWH3	(Frame Words)

1W	1W
SWH4	SPARE

SCI C

2W	1W	1W	8W	4W	4W	2W	1W	5W	4W	
SYNC	FRAME ID #	MAGN	LANG PROBE	IMS	UVS	IRR	SW	NMS	SPARE	(Frame Words)

SCI D

2W	1W	4W	16W	4W	5W	
SYNC	FRAME ID #	MAGN	UVS	SW	SPARE	(Frame Words)

Table 3-7 (continued)

SCI E

2W	1W	25W	1W	1W	1W	1W	(Frame Words)
SYNC	FRAME ID #	RADAR ALTIMETER DATA	RAH1	RAH2	RAH3	RAH4	

Probe Bus

SCI A (Cruise Science)

2W	1W	22W	2W	1W	1W	1W	1W	1W	(Frame Words)
SYNC	FRAME ID #	MAGN	MH1	MH2	MH3	MH4	MH5	SPARE	

SCI B (Encounter Science)

2W	1W	6W	5W	2W	2W	1W	1W	1W	1W	1W	1W	(Frame Words)
SYNC	FRAME ID #	NMS	IMS	LP	UVF	M	MH1	MH2	MH3	MH4	MH5	

1W	1W	1W	5W	(Frame Words)
IMSH1	IMSH2	IMSH3	SPARE	

TABLE 3-8. NOMINAL TELEMETRY MODES

Mode	Description	Rates, bps		Minor Frame Sequence
		Science Requirement	Nominal	
1	Probe bus engineering instruments	--	--	2 X { ENG A, B, C, D TCM A, B Spare, Spare
2	TCM	--	--	8 X TCMA, TCMB
3	Attitude determination	--	--	16 X TCMA
4	Cruise science	13	16	16 X SCI A
5	Engineering instruments - cruise science	13	16	Mode 1 decks alternated with mode 3 decks
6	Science encounter	220	2048	16 X SCI B
7	Computer memory recycle	--	--	16 X CMR
1	Orbiter engineering	--	--	2 X { ENG A, B, C, D TCMA B Spare, Spare
2	TCM	--	--	8 X TCMA, TCMB
3	Attitude determination	--	--	16 X TCMA
4	Cruise science	9	16	2 X SCI B, 7 X SCI A

Table 3-8 (continued)

Mode	Description	Rates, bps		Minor Frame Sequence
		Science Requirement	Nominal	
5	Engineering instruments - cruise science	9	16	Mode 1 decks alternated with mode 3 decks
6	Apoapsis	19	128 (32)*	SCI B, 4 X SCI D, ENG A, B, C, D TCM A, B 5 spares
7	Apoapsis - playback	19	128 (32)*	SCI B, 4 X SCI D, TCMA, B ENG A, B, C, D, five Stored Decks
8	Periapsis	86	128	SCI B, 15 X SCI C
9	Computer memory recycle	--	--	16 X CMR
10	Apoapsis	19	128/64 (64/32)*	SCI B, 7 X SCI D, ENG A, B, C, D TCM A, B, 4 spares
11	Apoapsis playback	19	128/64 (64/32)* (Playback = 64/32)	SCI B, 7 X SCI D, Eight stored decks ---

( )* = Effective science data rate

TABLE 3-9. DSN SUPPORT REQUIREMENTS – TELEMETRY

Function	Requirement	DSN Capability
Subcarrier frequency		
• Spacecraft	32, 768 kHz	20 Hz < capability < 45 kHz
Bit rates, bps		
• Spacecraft	8 to 2048	
Sequential decoding		
• Constraint length, bits	32	32
• Rate	0. 5	0. 5
• Tail, bits	≥ 20	8 to 48
• Frame length, bits	≤ 640	1200

Additional Studies

At the time of the midterm presentation specific analog conversion requirements had been established for the probe mission. These included 10 bit resolution for some instruments and 7 bit for others. However, for the probe bus and orbiter spacecraft no specific requirements were given and capability was established on the basis of existing hardware. This criteria was utilized to minimize cost, and hardware was found that provided conversion resolution to 6, 7, and 8 bits. This included hardware from Pioneer, Mariner, and the OSO Programs. Other criteria, which included cost control, science instrument accommodation, etc. determined the hardware selection process. These are discussed in Section 4; however, the OSO program hardware was selected which provides an 8 bit analog-to-digital conversion resolution. It was felt that there was no specific requirement for 8 bit resolution; however, some advantage might be gained from an 8 bit system due to ground computer compatibility and the possibility of increased scientific instrument accuracy which would perhaps minimize experiment hardware. The May 1973 science instrument update did specify 8 bit conversion accuracy for most scientific instruments on the probe bus and orbiter. Ten bit resolution was specified for two signals on the probe bus. Since this resolution requirement applies only to a single scientific instrument, and the existing OSO data handling hardware must handle all the science instrument's requirements, it is not planned to change the spacecraft to a ten bit system. Rather, the optimistic solution at this point is to require the instrument itself to perform ten bit conversions.

Data compression techniques were briefly examined but eliminated as a requirement due to increased cost, weight, and ground data processing complexity. To be a viable consideration the data compression would have to permit elimination of a power amplifier in the RF telecommunications subsystem and this was not possible for any of the multiprobe or orbiter missions.

Convolutional encoding techniques were also examined. At the time of the midterm, sequential decoding was required at the DSN due to hardware limitations. For this reason, the Pioneer 32 bit, quick look was selected for the baseline. Since the midterm, the decision to launch in the 1978 launch opportunity has permitted use of Viterbi decoding capability at the DSN. From the spacecraft standpoint, there is a weak preference for the Viterbi decodable class of convolutional codes. This is due to hardware simplicity associated with shorter constraint lengths and elimination of the tail sequence requirement.

3.4 PROBE DATA HANDLING REQUIREMENTS

This subsection describes the requirements for the probe data handling subsystems. Table 3-10 summarizes the data handling requirements for the large and small probes.

Table 3-11 summarizes the assignment of engineering and science telemetry data channels to telemetry formats on the large and small probes. These assignments are required to meet engineering and science data sampling requirements.

Derivation of Probe Data Handling Requirements

The mission unique requirement has been to maximize the amount of data that can be sampled and transmitted during probe entry and descent. A small amount of data storage is required during the blackout (ion sheath) portion of entry. During the early study phase, subsystem weight presented the most critical design factor, and data formats and rates were carefully and efficiently "tuned" to science payload requirements to eliminate unnecessary logic. Logic minimization has been a continuing goal, but (in view of the Atlas/Centaur launch vehicle selection) the forcing function has shifted from weight to cost considerations.

Each payload update for the probe missions has required establishment of new data formats and rates. It can be seen that these changes will continue until selection of the final payload. These changes, however, have been accommodated within the framework of the baseline design. This design allows flexibility in selection of input signal type (analog or serial-digital), data rates (simple countdown chain pickoff), or formats.

TABLE 3-10. PROBE DATA HANDLING REQUIREMENTS

Large probe

- Format, encode and modulate spacecraft and experiment data
- PCM/PSK/PM
- Convolutional encoding
- 276/184 bps data rate
- 3 descent data formats (plus 1 landed data format)
- 10 bit word size
- Engineering channels
  - 15 analog; 2 serial digital; 19 discrete
- Experiment channels
  - 58 analog
  - 4 serial digital; 9 discrete
- 4096 bits data storage

Small probe

- Format, encode and modulate spacecraft and experiment data
- PCM/PSK/PM
- Convolutional encoding
- 15 bps data rate
- 2 data formats
- 10 bit word size
- Engineering channels - 13 analog; 2 serial digital; 19 discrete
- Experiment channels
  - 12 analog
  - 2 serial digital
- 480 bits data storage

TABLE 3-11. PROBE TELEMETRY FORMAT ASSIGNMENTS

Large Probe

Descent I and II

3 Words	2W	22W	41W	17W	5W	3W	2W	33W	(Frame Words)
Sync + ID	Temp S	Mass Spect	Cloud Part	AU Ext	Nep	Hygro	Play Back	Sub-Com	

Descent I (Sub Com Frame)

1W	1W	4W	4W	3W	20W	20W	2W	19W	2W	2W	2W	(Frame Words)
Sync ID	Temp	ACC	Press S	Tur	Solar Flux	Planet Flux	Trans	Engr	Nep	Mass Spect	Play Back	

6W	33W	10W	3W
Sci House	Cloud Part	Aur Ext	Nep

Descent II (Sub Com Frame)

	44W	(Frame Words)
SAME AS DESCENT I SUB COM	Mass Spect	

Table 3-11 (continued)

Landed and Store 2

2W	30W	
Sync + ID	Accelerometer Data	(Frame Words)

Store 1

2W	17W	9W	4W	
Sync + ID	Shock Layer Rad	Accelerometer	Engr	(Frame Words)

Small Probe

Store

2W	30W	
Sync + ID	Accelerometer Data	(Frame Words)

Descent

2W	9W	5W	4W	3W	3W	2W	2W	6W	9W	
Sync + ID	Nep	Accel	Magn	Temp	Press	Osc	Play Back	Engr	Spares	(Frame Words)

TABLE 3-12. LARGE PROBE SAMPLING RATES

Parameter	Resolution Required	Derived SPS Requirement	BPS Above 20 km: Implemented (10 bit words)	BPS Above 20 km: 7 and 10 Bit Words
Accelerometer (axial)	10	1	. 1. 6	. 1. 6
Accelerometer (lateral)	10	0. 05	0. 5	0. 5
Turbulence	7	0. 143	1. 6	1. 1
Transponder	10	0. 033	1. 1	1. 1
Mass spectrometer	*	0. 0032	48. 5	48. 5
Temperature	10	0. 18	4. 3	4. 3
Pressure	10	0. 09	2. 2	2. 2
Cloud particle	*	0. 44	106. 2	106. 2
Solar flux	10	0. 07	10. 8	10. 8
Planet flux	10	0. 07	10. 8	10. 8
Aureole	*	0. 65	42. 0	42. 0
Nephelometer	*	0. 65	13. 5	13. 5
Hydrometer	7	0. 18	6. 5	3. 78
Playback	D	0. 5	5. 4	5. 4
Engineering and science housekeeping	7		14. 0	9. 8
Synchronization and instrument detection	*		7. 0	7. 0
			276. 0	268. 58

*Digital data

There are specific goals in format design which are useful in hardware minimization. First, the format should be short as possible. This, however, can lead to inefficient use of the data bandwidth. For example, if the data format were 32 words long the minimum signal sampling rate would be 1/32 times the bit rate. If the bit rate were 64 bps, 2 bps would be the minimum sampling rate that could be applied to a signal. If the bandwidth is used inefficiently, then for a given science instrument requirement a higher data rate would be required from the probe. This higher rate, of course, affects the rf link design. So a logical tradeoff exists between more logic for format design and more power for the rf subsystem. In actual fact (like the spacecraft), the rf power output capability is not continuously variable, but exists at discrete levels due to addition or deletion of power amplifiers. So the design goal has been to develop formats that make most efficient use of a particular rf capability.

Second, due to the nature of available logic components, it is desirable to develop formats with lengths in powers of two. For example, formats of 32 or 64 words are desirable. The Thor/Delta and Atlas/Centaur baseline designs differ with respect to data formats and rates due to differing descent characteristics and a payload update mentioned previously. The present Atlas/Centaur baseline design uses a 64 word format for both the minor frame and the subcommutator.

Additionally, logic saving assumptions were made during establishment of the probe data handling subsystem requirements. These included utilization of a single ten bit word format and utilization of the data memory only during the high deceleration phase.

Certain of the probe scientific instruments require ten bit (or 0.1 percent) analog-to-digital encoding accuracy. Other data, for example the probe and science instrument housekeeping data, require only 7 or 8 bit (0.5 percent) accuracy. For purposes of design simplification, the differing requirements were ignored and a simple 10 bit telemetry word was used across the board. Clearly there is some loss in data efficiency associated with this approach and some telemetry bits could be saved if seven bit words were transmitted as seven rather than 10 bits. Actually, the savings are small due to infrequent sampling of the seven bit words.

The bit savings were analyzed for the large probe 276 bps descent data format. This format was chosen for analysis as it is the only format in which the science data requirement closely approaches the data transmission capability of the rf link.

Three science instruments, all science housekeeping, and all engineering data channels (as shown in Table 3-12) could be reduced from 10 to 7 bit words. If this were done, an additional 2.69 percent data bandwidth would be available for expansion. This is equivalent to approximately 7.42 bps or 13.8 words in the 512 word major frame.

Increased complexity would be required in the probe data system in order to sample and transmit both 7 and 10 bit words. The cost impact of such a change, however, would not be excessive (less than $100K). The same analog-to-digital converter would be utilized with only the 7 MSBs utilized for 7 bit words. Changes would be made to the formatter to tag each word for 7 or 10 bit transmission.

It was, therefore, concluded that the cost and weight expenditures to implement both 7 and 10 bit word compatibility were not worth this small savings in data bandwidth.

The probe data storage is used only for storing data during the high deceleration phase of entry. At this phase data rates are relatively high and rf communications are blacked-out, thereby requiring storage. An additional use of data storage would be as a data buffer. The buffer would store data during high instrument activity periods and playback at low activity periods. This use is attractive from the standpoint that data accumulation is desired at equal densities at all altitudes. Since the descent velocity is monotonically decreasing from entry through parachute release and again through landing, data could be stored during high descent rates and played back during slower descent rates.

The data buffer concept was ruled out early in the study phase due to:

1) Increased circuit complexity and associated costs

2) Higher risk in data recovery due to the requirement to playback data on lower descent rates (i.e., nearer the surface)

3) Inability to reduce the size of the rf power amplifier and still meet the science data requirements

4. SUBSYSTEM TRADE STUDIES

In the search for optimum design configurations and suitable hardware for the command and data handling subsystems, a number of major tradeoff studies were performed. These trade studies were concerned with both the probe bus and orbiter spacecraft, and also with the large and small probes. Although each of the studies were directed toward specific design objectives and employed particular criteria for their evaluation, they were generally intended to establish optimum designs that would both meet the following fundamental design objectives while still meeting the specified subsystem requirements:

1) Minimize the development risk and provide a low cost implementation through maximum use of existing, available space hardware and by maintaining a maximum degree of commonality (both units and circuits) between both the probe bus and orbiter spacecraft and the probes.

2) Maximize equipment reliability through use of reliable components and by design simplification.

3) Achieve minimum mass, power, and volume.

4) Where possible, maintain design flexibility to accommodate changes in subsystem and science payload requirements as development progresses.

The major aspects of each of these studies are discussed in this section as noted below.

Command subsystem studies are presented in subsection 4.1; they include:

1) Command interface methods.

2) Prevention of spacecraft inadvertent-irreversible command execution.

3) Spacecraft command storage analysis.

4) Probe cruise timer.

5) Receiver reverse unit.

Data handling subsystem studies are described in subsection 4.2 and include:

1) Telemetry data recovery analysis.

2) Data handling interface methods.

3) Orbiter data storage analysis.

4) Probe data storage hardware.

5) Probe stored data playback techniques.

6) Probe multiple data formats.

7) Probe multiple data rates.

A survey of available subsystem hardware is discussed in subsection 4.3. This subsection presents the results of a search to locate appropriate command and telemetry data handling subsystem hardware that has already been developed for previous space programs. The survey included prior developments by both Hughes and by other space equipment manufacturers.

Subsection 4.4 contains a summary of a tradeoff made between alternate pyrotechnic configurations to determine the best designs for the orbiter, probe bus, and probes. The analysis was based on initiator (bridge wire and hot wire) characteristics, reliable power switching methods, and power source characteristics. Selection was made of configurations which are highly reliable, low in cost, and that meet system requirements.

Finally, subsection 4.5 presents the results and conclusions of a study that investigated the use of a programmable on-board data processor for performing major portions of the spacecraft command and data handling functions. A design that employs a small general purpose computer, on a time-shared basis, is described and compared with alternate designs. Major topics discussed in this section are:

1) Centralized command and telemetry data handling subsystem

2) Computer hardware survey

The midterm Thor/Delta performance requirements were generally utilized as the parameters for these trade studies. For the sake of completeness, studies that have been performed since the midterm review, have been included. In some cases these newer studies rely upon and include current Atlas/Centaur requirements. Where this is true, the study includes a statement indicating that it is based upon current Atlas/Centaur data.

General effects to the trade studies of the current Atlas/Centaur mission are described in Section 6.2.

4.1 COMMAND SUBSYSTEM STUDIES

Various aspects of the command subsystem were studied in order to establish optimum design techniques and methods of implementation. The studies emphasized utilization of existing design techniques and available space hardware so as to minimize development risk and implementation costs. Additional tradeoff objectives were to provide adequate safety margins and to achieve minimum mass, power, and volume.

The specific objectives, results, and conclusions of these studies are summarized below.

1) Command Interface Methods. A study was performed to define a command interface that would satisfy all science and engineering requirements. After considering various design techniques, the method of command interface that was selected is the same as that used on the OSO-I program. Since the command interface for OSO was developed specifically for use on a scientific spacecraft, it is particularly well suited for use on Pioneer Venus. In addition to meeting all of the functional requirements, it meets all of the previously stated design objectives.

2) Prevention of Spacecraft Inadvertent-Irreversible Command Execution. A survey was conducted to find acceptable means for preventing the inadvertent execution of irreversible commands. Six basic techniques were investigated. They included techniques employed in previous space missions such as Surveyor, Pioneer F-G, OSO, and others. The use of interlocked commands was selected as the preferable method for Pioneer Venus. This method uses two independent, contiguous commands employing two independent Hamming code words. Since each Hamming code word is double-error detecting, an interlocked command cannot be inadvertently executed unless it contains five or more errors. Also, since this method was used on earlier Pioneer missions, it minimizes changes and impact on ground processing hardware and software.

3) Spacecraft Command Storage Analysis. Requirements for command storage on both spacecraft are primarily derived from the need to automatically reorient the spacecraft and to deploy booms as soon as possible after spacecraft separation from the launch vehicle and, for the orbiter, to execute the periapsis

science instrument sequence. A study was performed to select an appropriate storage media for storing up to 64 commands on the probe bus and orbiter spacecraft. Both random access memories and shift registers of various semiconductor technologies were considered. The study concluded that use of a long MOS shift register was preferred, since a sequence of time delayed commands is sufficient to implement the command storage requirements. This method was selected primarily because it minimizes the amount of hardware required for implementation.

4) Probe Cruise Timer. Another tradeoff was performed between alternate designs of the probes' cruise timer in order to determine an approach which would consume minimum power during the cruise period, meet the timer accuracy requirement, and provide minimum technical risk. Two alternatives to the original design were considered. The selected approach uses the original oscillator/timer, but employs a switching regulator to provide more efficient voltage regulation than the series regulator proposed for the original design. This results in a reduction of total cruise power from 60 to 30 mW.

5) Receiver Reverse Unit. Alternate designs of the spacecraft receiver reverse unit were analyzed to determine the best method to control the omniantenna/receiver combination to assure acquisition of an operating receiver/demodulator when only one omniantenna has earth visibility. The selected unit is a simple two state device.

Each of these studies is described in more detail in the following sections.

Command Interface Methods

The purpose of this study was to define an optimum command interfacing method for controlling of science instruments and engineering equipment on the four Pioneer Venus space vehicles. The particular interface to be investigated was that between the command subsystems and the users. A user is defined as a receiver of any command; the receiver may be a science instrument or another engineering subsystem. The study was to include any commands that are to be transmitted from the command subsystem to a user, either in real time or from a stored command memory.

The investigation was guided by a procedural approach, which was designed to meet the fundamental objectives previously stated (i.e., use available space hardware to minimize cost and development risk; achieve maximum reliability and hardware commonality; minimize mass, power, and volume; where possible, maintain flexibility to accommodate changes). The procedure for conducting the study was:

1) Establish guidelines that define optimum command interface features and characteristics for scientific type spacecraft;

the interfaces must be sufficiently flexible to accommodate a multiplicity of science instrument and equipment complements with maximum standardization.

2) Survey and evaluate existing hardware designs; consider new designs only if appropriate existing hardware is not available.

3) Select the most appropriate hardware design based upon the above objectives and guidelines.

4) Define the specific characteristics of the selected design and establish their particular relationship with the Pioneer Venus applications.

After the interface guidelines were established, a survey was conducted to solicit information on previously developed command subsystem hardware from known suppliers of space equipment. Details of this survey are reported in subsection 4.3. Evaluation of existing equipment designs from six different companies resulted in selection of the command interface hardware designed for the OSO-I program.

The OSO interface hardware is felt to be an optimum choice for all four space vehicles. After all the surveyed equipment was evaluated, it was found that most of the other equipment designs met portions of the above objectives and guidelines to varying degrees. However, the OSO hardware was the only interface equipment located that meets all of them without modification. This hardware is a very current design and uses modern LSI technology to reduce mass and improve reliability. It was developed specifically for use on scientific spacecraft. It is particularly well suited for Pioneer Venus, since the interface requirements for these scientific spacecraft are very similar to those of OSO-I.

A summary of some of the more pertinent aspects of this study is given below. Detailed results of the study may be found in Reference 4-1.

Interface Guidelines

Guidelines that define optimum command interface features and characteristics are summarized below. Whereas these guidelines were specifically established for this study, they have evolved from Hughes extensive experience in spacecraft design (e.g., Surveyor, ATS, OSO, plus several military and communications satellites) and from the experience of other agencies.

1) Design modularization should be employed to achieve configuration flexibility. Modularization will allow tailoring the number of hardware modules to accommodate a higher or lower number of command interfaces as dictated by the specific application.

2) The command interface should be capable of providing two basic types of digital commands to users. The first is a discrete, pulse type command to initiate modes and events; the second is a serial data or magnitude type command to transfer digital data and quantitative information to users.

3) A logical 1 should be associated with a source of energy (a positive voltage source supplying current, or a negative voltage sink absorbing current).

4) A logical 0 should be an inert state (such as zero volts with no current flow) such that the control wire could be shorted to ground or open circuited without creating a false signal. An inert state also permits a considerable spacecraft power saving, since a unit can be completely powered off when it is inactive.

5) Commands or control signals from redundant sources should be able to be "OR'ed" together, and a single component failure in one source should not prevent proper operation of the redundant source. This is achieved by providing a diode "OR"ing capability in the command interface receiving buffer. This feature also prevents an open wire, an open connector pin, or a short to ground in the harness from disabling the redundant command source or path. The inactive unit should be powered off in a standby redundant mode of operation. This is possible with the logic levels defined above.

6) A command or control line should have a fanout of at least two (when using a standard input buffer), enabling a single output to control two functions simultaneously.

7) System noise immunity must be achieved. A good way to achieve this is by providing threshold detection, which has hysteresis and a large signal/threshold ratio, plus filtering (e. g. , a relatively large signal voltage swing around the hysteresis threshold levels plus a controlled integration of current over time).

8) A command should be a logic signal rather than a power signal to minimize EMI noise generation.

OSO Interface Design Description

The command output modules of the OSO command subsystem meet all of the command interface guidelines described above; in OSO terminology, these modules are called remote decoders. Each of these modules provides the interface capability for 64 pulse commands and 4 serial data (or serial magnitude) commands.

Tables 4-1 and 4-2 summarize the basic features and output characteristics of pulse and serial magnitude commands, respectively. Each pulse command is transmitted to the user on a single line, while each magnitude command is transmitted via three interface lines; one line is for transmission of serial NRZ data, one is for the associated read clock signal, and the third line is for the associated read envelope signal that defines the time period during which valid data is to be recognized.

Interface Circuit Descriptions

For user convenience, two types of integrated circuits have been developed by Hughes to alleviate any potential interfacing difficulties. They are compatible with the OSO command hardware, and were developed to permit standardization of signal interfaces between the command subsystem and the users. Furthermore, they can accommodate buffering requirements for essentially any type of low frequency digital signals. It is recommended that these circuits be employed in the users equipment to insure interface compatibility with the command subsystem. Otherwise, the interface circuits in the users equipment should have equivalent characteristics. These integrated circuits are:

1) Standard Input Buffer. This device, Hughes part number 908974, is a dual line receiver that is used to receive command signals.

2) Standard Output Buffer. This device, Hughes part number 908973, is a quadruple line driver. It is recommended that this circuit, or its equivalent, be employed by the user to provide

TABLE 4-1. PULSE COMMAND CHARACTERISTICS

Parameters	Characteristics
Logic 1 (execute)	≥12 V while supplying 4 mA (open circuit voltage = +15 ±2 V)
Short circuit protection	Current limiting is provided for protection against shorts to ground
Fanout capability	Two (with standard buffers)
Logic 0 (quiescent)	0 V (signal ground) through a source impedance of 5.3 ±2.9 kΩ
Pulse width	50 ms nominal
Voltage rise time (10 to 90 percent)	2.5 μsec maximum (with load of 700 pF capacitance in parallel with 30 kΩ resistor to ground)
Voltage fall time (90 to 10 percent)	15 μsec maximum (with load of 700 pF capacitance in parallel with 30 kΩ resistor to ground)

TABLE 4-2. SERIAL MAGNITUDE COMMAND CHARACTERISTICS

Parameter	Control Line Characteristics		
	NRZ Data	Read Clock*	Read Envelope*
Logic 1	≥12 V while supplying 4 mA (open circuit voltage = +15 ±2 V)	Same	Same
Short circuit protection	Current limiting is provided for protection against shorts to ground	Same	Same
Fanout capability	Two (with standard input buffer Hughes Part No. 908974	Same	Same
Logic 0	0 V (signal ground) through 5.3 ±2.9 kΩ source impedance	Same	Same
Voltage rise time	Less than 2.5 μsec	Same	Same
Voltage fall time	Less than 14 μsec	Same	Same
Load for rise and fall times	700 pF in parallel with a standard input buffer (or 30 kΩ). This corresponds to a total line length ≤14 ft	Same	Same
Word length	10 bits		
Bit rate	16 kbps ± 1 percent	Same	—
Envelope pulse width	—	—	0.680 ± (TBD) ms

*Characteristics of the read clock and read envelope lines are the same as for the NRZ data line, except as noted.

any necessary event type signals; occasionally, these types of signals are used to notify the command subsystem of an occurrence that requires command action.

A major feature of these integrated circuits is their exceptionally good immunity to noise. Since the driver and receiver circuits have been designed to be compatible, they eliminate many interface problems related to electrical ground references; e. g. , connection transitions between signal ground, power ground, etc. associated with the interfacing equipments. They also insure proper matching of electrical interface characteristics, including cabling capacitance. In addition, they provide simple and reliable methods for cross-strapping of redundant units.

The 908973 and 908974 interfacing circuits may be used in a variety of modes, depending upon the specific interfacing requirements; detailed descriptions of their usage for various interfacing applications may be found in References 4-1 and 4-2.

Spacecraft Command Applications

The command subsystems of the probe bus and orbiter spacecraft provide the means for commanding science instruments, other subsystems, and pyrotechnic devices. Commands can be transmitted either in real time, via the command uplink, or from the stored command memory. Both pulse and serial magnitude types of commands are utilized. Spacecraft commands are distributed by redundant sets of remote decoder modules. Because of this redundancy, any single failure in a remote decoder will not be detrimental to the user's operation. Each decoder is capable of distributing 64 discrete pulse commands and 4 serial magnitude commands. The pulse commands are low power signals with a nominal logical 1 time duration of 50 ms. Each serial data command channel consists of three low power signals, whose logical 1 and logical 0 states are time dependent such that the intelligence (magnitude data) is determined by the relative timing of the 1 and 0 states. The time duration for the transfer of one serial data magnitude command is nominally 1 ms. Command subsystem/user interface signals from redundant decoders are "OR'ed" together at the user's end in order to provide a redundant command interface.

Probe Command Applications

The command sections of the probe command and data handling subsystems receive pulse and magnitude commands from the probe bus prior to separation of the two vehicles. These commands are used to initiate and terminate preseparation tests, to provide 20 bits of time information to the probe's cruise timer, and to start the timer.

After separation of the probes from the probe bus, the timer initiates preentry events with an accuracy of ±20 sec following a nominal 20 day cruise period. After the separation, all pulse commands generated by the command section result from fixed stored commands and from event occurrences signaled by probe subsystems and science instruments. Only pulse type commands are provided by the probe command section; no magnitude type commands are required. The pulse command characteristics for the probes are the same as for the spacecraft.

Each probe must process a small number of event signals. These signals originate as outputs from various sensors and/or science instruments. The electrical output characteristics of these signals shall be compatible with the input requirements of the standard input buffer interface circuit (Hughes part number 908974); these required output characteristics are shown in Table 4-3. To insure this compatibility, it is recommended that the standard output buffer (Hughes part number 908973) be used.

Prevention of Spacecraft Inadvertent-Irreversible Command Execution

Summary

A major concern with any spacecraft mission is protecting against the possibility of any event that may jeopardize the total mission and that cannot be corrected from the ground. This could be the case for certain critical functions that may erroneously be executed and that are irreversible; i. e., where no corrective action can be taken to save the mission. An inadvertent command to fire a squib driver is an example.

TABLE 4-3. PROBE EVENT SIGNAL REQUIREMENTS

Parameter	Characteristics
Logic 1 (execute)	+12 to +17 V while supplying 4 mA
Short circuit protection	Current limiting to 50 mA. for protection against shorts to ground
Fanout capability	Two (with standard buffers)
Logic 0 (quiescent)	0 V (signal ground) through a source impedance of <30 kΩ
Voltage rise time (10 to 90 percent)	25 μsec maximum (with load of 700 pF capacitance in parallel with 30 kΩ resistor to ground)
Voltage fall time (90 to 10 percent)	100 μsec maximum (with load of 700 pF capacitance in parallel with 30 kΩ resistor to ground)

Because of this concern, a study was conducted to select a technique for preventing inadventent irreversible commands from being executed by the probe bus/orbiter command subsystem. The first part of the study consisted of a survey of prevention techniques that have been used, or suggested for use, in previous space missions. These techniques include various error detecting codes, address check, repetition of the data, interlocked commands, and command verification via telemetry.

The second part of the study task consisted of a tradeoff to select the technique best suited to the Pioneer Venus requirements. Interlocked commands, used by ARC on earlier Pioneer missions, were selected as best for Pioneer Venus. Interlocked commands, to be issued only from a command memory for mission critical functions, were selected for the following reasons:

1) Flexibility in handling both irreversible and noncritical commands with a single format

2) Compatibility with existing Pioneer ground equipment

3) Protection against human error in initiating an irreversible command, since two independent actions are required for initiation

Command verification via telemetry readout of magnitude registers was recommended as a backup technique for verifying critical magnitude commands.

Discussion.

The study established that the maximum number of errors that must be detectable is approximately four. Four was chosen for Pioneer Venus, since it guarantees very low probability of accepting an invalid command, and because it is consistent with previous spacecraft command subsystem designs.

The survey of techniques included those used by Pioneer F and G, Surveyor, OSO, and other space missions. The techniques investigated can be classified roughly as follows:

1) Interlocked commands. Interlocked commands are defined as a sequence consisting of two independent commands, each command containing a block code that is double error detecting. An interlocked command cannot be inadvertently executed unless it contains five or more errors. The interlocked command concept is easily adapted to noncritical commands by simply omitting one of the two independent commands; no change in the command format is necessary. This is the preferred method.

2) Block codes. A block code is a group of bits, added onto the command word, that contain parity information for detecting multiple errors. Codes investigated were Hamming and cyclic codes, including BCH. A single block code word is adequate for noncritical commands.

3) Transmission of the command and its complement. This method is very inefficient, in terms of transmission rate, and does not guarantee that all double errors will be detected. This method was discarded for these reasons.

4) Command format check. A command format check consists of verifying that the received command contains an error-free address sequence. This is an effective technique for detecting burst errors, but is inadequate in a link where errors tend to be independent. This technique was not selected since it does not provide enough error detecting capability.

5) Ground verification before execution. This method requires an additional command to initiate execution after the command has been verified. In effect, two transmissions are required to execute a single real time command. Because of the built-in time delay, this technique was eliminated from consideration for irreversible commands.

The detailed rationale for selection of the interlocked command method may be found in Reference 4-3.

Spacecraft Command Storage Analysis

Requirements Analysis

The requirement for command storage on both spacecraft is derived from the need to automatically reorient the spacecraft and deploy booms as soon as possible after spacecraft separation from the launch vehicle. Additionally, orbiter events of orbit insertion during occultation and occulted periapsis operations require stored command sequencing.

Command sequences for control of events following separation from the launch vehicle will be entered into the command memory and their correctness verified prior to launch. Upon separation command memory operation will be begun by sensing the separation switch closure and subsequently enabling the spacecraft stored command processor. All other stored sequences will be initiated by the first edge of the first 30 min clock interval pulse which follows reception of a real time command to execute memory operation. By this means, the command to initiate the stored sequence can be sent more than once to increase the probability of its successful execution, and the time criticality of its transmission is eliminated.

Memory words consist of either time information, command information, or zeros. Upon initiation,the stored command processor will increment the memory, which is in the form of a 2048 bit shift register, until it encounters a non-zero bit that indicates time or command information. If the first word is a command word the indicated command will be immediately executed as will subsequent command words until a time word is encountered. The time information will be entered into the stored command processor time counter which will then be decremented by the spacecraft master clock until it underflows. Upon sensing the underflow condition the stored command processor will interrogate the memory and execute commands until it encounters the next time word. The process continues and the memory end-around-shifts until a command to halt memory execution is encountered either in the real time command link or in the command memory. The delta time delay technique indicated simplifies the determination of command sequences on the ground and reduces time register size requirements on the spacecraft. The end-around-shift feature allows cyclic or noncyclic sequences to be executed by either omitting or including in the sequence a command to halt memory operation.

Command word size is determined by the magnitude command storage requirement of 16 bits plus 2 bits to indicate time/magnitude/pulse command word type plus 5 bits for addressing for a total of 23 bits minimum. Time information requires the 2 bit word identifier plus 16 bits of time information. Pulse command information requires the 2 bit word identifier plus a 9 bit address. A 24 bit word is selected since a smaller word would require transmitting more than one command to load a magnitude command word thereby unduly increasing the time required to load the command memory.

Stored sequence requirements at launch, orbit insertion, and during on orbit occultations determine command memory size in words. Table 4-4 presents sequences for these events. Using a 2^N size shift register it is seen that a 1024 bit (42 word) register is sufficient. A 2048 bit register is selected to accommodate growth.

Timing of the radar altimeter rf pulse relative to the sun pulse depends on spacecraft-Venus geometry as does the altimeter antenna elevation angle which is slewed using a phased array approach. The desire is to transmit the rf pulse when the antenna is pointed along the local vertical. It was found that a pulse timing and antenna pointing algorithm exists and requires approximately six constants to be entered before each periapsis pass. It was assumed that the algorithm would be implemented within the experiment. Therefore, the periapsis occultation sequence in Table 4-4 indicates the need to extract six constants, in the form of magnitude commands, from the command memory.

Hardware Implementation Analysis

A study was performed to determine the most applicable method for implementation of the probe bus and orbiter command memories. Technologies and techniques were both studied. The results of the study led to

TABLE 4-4. BUS COMMAND MEMORY SEQUENCES

Subsequence	Time	Command	Memory Words: Time	Memory Words: Command
Post separation	Sep +0	Separation switch initiates memory operation	–	–
	Sep +10 sec	JCE-ON	1	1
		Load despin data		2
		Execute despin		2
	Sep +50 sec	Terminate despin	1	1
		PCU-ON		1
		Deploy booms		2
		PCU-OFF		1
	Sep +23 min	Load reorientation data	1	3
		Execute reorientation		2
	Sep +26 min, 20 sec	Terminate reorientation	1	1
		Total words:	4	16
Orbit insertion	I -45 min	Initiate memory operation by ground command	–	–
	I -40 min	Test command	1	1
	I -25 min	Test command	1	1
	I-200 sec	Execute liquid thrusting	1	2
	I +0	Terminate liquid thrusting	1	1
		PCU-ON		1
		Ignite OIM		2
		PCU-OFF		1
	I +1h	Load reorientation	1	3
		Execute reorientation		2
		Terminate reorientation	1	1
	I +1h, 20 min	Load MDA despin data	1	2
		Despin MDA		1
		Total words:	7	18
Periapsis occultation	P -15 min	Initiate memory operation by ground command	–	–
	P -8.4 min	1000 km science ON	1	2
	P -4.2 min	Altimeter rate change	1	1
	P -0	Change altimeter algorithm	1	6
	P +4.2 min	Altimeter rate change	1	1
	P +8.4 min	1000 km science OFF	1	2
	P +11 min	Fire jets for period trim	1	2
	P +11.2 min	Terminate jet fire	1	1
		Total words:	7	15

the selection of a static MOS shift register for command storage. This selection was based on its simplicity of configuration and the fact that only 64 delay functions are required to be implemented.

The methods reviewed were shift registers and random access memories (RAM). The hardware reviewed consisted of core RAM, MOS shift registers, and MOS RAM. The advantages and disadvantages of each method and hardware type are briefly summarized below.

Random Access Memories. The advantages of using a RAM are as follows:

1) Any memory location can be loaded independently of other locations. This feature allows memory corrections to be made in a minimum time.

2) Faulty sections of the memory may be deleted by a jump command.

The disadvantages of using a RAM for a command memory are:

1) Extra logic is required for keeping track of word locations; this hardware basically consists of a sequencer and address register.

2) Complexity of the RAM itself is usually high because of multiple drivers and sense amplifiers.

Some characteristics of magnetic and semiconductor RAM memories are as follows:

1) Magnetic memories are nonvolatile and can be power strobed to keep power to a minimum. They are usually large and, unless the required memory is greater than 2×10^5 bits, they are usually not preferred unless power is the prime consideration.

2) MOS semiconductor RAMs are the preferrable semiconductor technology for this application. The reasons for preferring MOS devices are that they represent the highest bit density per package and the lowest power per bit as compared with other existing semiconductor technologies. The primary advantage of MOS RAMs over magnetic RAMs is that they can be easily configured to a particular bits/word combination. The primary disadvantage is that they are volatile and must be powered continuously.

Shift Registers. The advantages of using a shift register for a command memory are as follows:

1) A minimum of control hardware is required.

2) Memory length can be added easily.

3) Recycling is extremely simple.

The disadvantages of using a shift register are:

1) An entire stored command sequence must be loaded with no errors; e.g., if the bit error rate is 10^{-5} and sixty-four 36-bit commands must be loaded, then the probability of loading correctly is 97.5 percent.

2) If one bit in the memory fails, the entire memory is lost.

3) Only delay functions can be implemented.

4) They are volatile and must be powered continuously.

Probe Cruise Timer

A tradeoff was performed between alternate designs of the probes' cruise timer in order to determine an approach which would consume minimum power during the cruise period, meet the timer accuracy requirement, and provide minimum technical risk. The primary goal was that reduction of power consumption ultimately allows a reduction in the mass of the probe batteries. Three different designs are considered and compared. They employ various combinations of frequency sources and voltage regulation methods and range from 15 to 60 mW of power. The selected approach uses a 1 MHz crystal oscillator in conjunction with a switching regulator and consumes 30 mW.

Background

This study was undertaken subsequent to the midterm review to determine alternate means of reducing power consumption in the probes' cruise timer and the relative cost and complexity of those alternatives. The overall accuracy requirement for the timer is ±20 sec in 20 days, including oscillator stability plus timer resolution. Assuming ±2 sec resolution for the timer, oscillator stability and aging effects must be kept to approximately 1×10^{-5} (17.3 sec in 20 days).

The initial design employs a 1 MHz oscillator, complementary MOS (CMOS) logic for the countdown chain and timer, and a series regulator to provide a +5 V logic supply from the 18 V probe battery source. This design requires 60 mW of power during the 20 to 23 day cruise period prior to entry into the Venus atmosphere. Most of the power consumed is in the timer's oscillator, the initial few stages of countdown, and in regulation inefficiency; the latter is due to the fact that the timer operates directly from the probe battery which must be regulated down from approximately 18 to 5 V for use by the timer.

Sixty mW of power represents 28.8 W-hr of energy over a 20-day cruise period. At the present battery energy density, this is equivalent to approximately 0.45 kg (1 lb) of battery weight; a 30 mW reduction in timer power would reduce the total weight of each probe by 0.22 kg (0.5 lb).

Cruise Timer Design Alternatives

The study was concentrated upon the two primary sources of power consumption: the oscillator and the voltage regulator. Low frequency timing references were investigated, as low frequency oscillators in general require less power than higher frequency circuits. Alternate methods of regulation were considered, as was operation directly from the probe battery voltage.

A summary of the timing references considered, with relevant design implications, is given in Table 4-5. The "G-T cut" quartz crystal is generally undesirable for space use due to its physical size and questionable ability to withstand the mechanical stresses of a normal spacecraft launch. The bimetallic tuning fork, in addition to being bulky, cannot provide the accuracy required of the cruise timer. The quartz tuning fork reference will not provide the 1×10^{-5} stability over the required temperature range; however, the stability can be controlled by controlling the temperature of the crystal or by temperature compensating the oscillator circuit, thereby achieving the desired accuracy. The initial design uses an "A-T cut" quartz crystal and has the advantage that it inherently meets the required accuracy, it uses a space proven design, and it uses presently qualified parts. The single disadvantage is the higher power dissipation.

The greater part of the power dissipation in the initial design (about 75 percent) is attributable to series regulation for conversion of the 18 V probe battery supply to the 5 V used by the timer. The use of a switching regulator, in lieu of the inefficient series type, can reduce the total power required by at least 50 percent (to 30 mW) at the cost of a slight increase in parts. The switching regulator would not be advantageous in a design using a low frequency crystal, since the current drain for this approach is quite small (less than 1 mA) and the efficiency of the switching regulator at extremely small loads is not significantly different from that of the series regulator.

Results

The results of the study indicate that two alternate approaches are worth consideration. The first uses the initial oscillator design but with a switching regulator to improve regulation efficiency; the second employs

TABLE 4-5. CRUISE TIMER TIMING REFERENCES

Timing Reference	Manufacturer	Space Qualified	Frequency	Timer Stability + Aging(1)	Timer Power, mW
"A-T cut" quartz crystal	Bliley	Yes	1 MHz	1×10^{-5}	30 (2)
"G-T cut" quartz crystal	Bliley	No	250 kHz	1×10^{-5}	20 (2)
Quartz tuning fork	Statek	No	32 kHz	5×10^{-5}	15 (3)
Bimetallic tuning fork	Bulova	No	16 kHz	1×10^{-4}	15 (3)

(1) Stability specified as $\frac{\Delta f}{f}$ over temperature range from 0 to 40°C; aging specified as $\frac{\Delta f}{f}$ for 1 year.

(2) Includes switching regulator inefficiency at 18.2 V bus.

(3) Includes series regulator inefficiency at 18.2 V bus.

a low frequency quartz tuning fork which is much lower in power than the initial design and retains the series regulator of the initial approach.

The first approach is preferred in spite of the fact that it consumes 30 mW compared to 15 mW for the latter. This is because the initial oscillator is a proven design and requires no new part qualifications. The latter approach uses a quartz tuning fork timing reference that inherently is not as stable as the initial design. The cost of designing around the stability (e.g., by temperature control) is enough so as to reduce the attractiveness of this approach. In addition, the device is not presently qualified for use in space and the manufacturer has not supplied parts to Hughes for space qualification in the past.

Receiver Reverse Unit

When only one omniantenna has earth visibility, acquisition of an operating receiver/demodulator is vital; otherwise this will result in the inability to command the spacecraft. The function of the receiver reverse unit (RRU) is to control the omniantenna/receiver combination, either automatically or by command.

Two RRU functional designs have been originated and analyzed. The first is essentially a two state device which issues standard commands to the transfer switch between the omniantennas and the receivers altering of antenna/receiver configuration. Switching would not be necessary if both omniantennas could be permanently connected to both receivers, but such a permanent connection results in fringes in the resulting omniantenna pattern. On the orbiter, command reception is not possible at extreme ranges using the omniantennas and the DSN 26 meter net so a switch was incorporated to allow use of the high gain mechanically despun antenna (MDA).

Factors considered in determining the RRU switching algorithm are the following:

1) The switch controls downlink energy to the antenna subsystem as well as uplink energy. Thus, switching of the SPDT switch when unwarranted can cause unnecessary loss of downlink.

2) Both switches are electromechanical and thus have limited cycle lifetimes.

3) Periods of time will exist during which no uplink energy will be received due to ground station outages, occultations, addressing of the alternate Pioneer Venus spacecraft, or ground station utilization conflicts with other projects.

4) Spacecraft attitude can be such that only one omni has earth visibility.

5) Single unit failures shall not be capable of disabling the ability to command the spacecraft.

6) Receivers and spacecraft are frequency addressable.

The following algorithm results from considering spacecraft safety as the prime algorithm selection criterion.

1) Upon sensing that no valid real time command has been received within the preceding interval, or that no command has been issued from the stored command processor to force the state of the antenna/receiver transfer switch within the preceding TBD hours, the RRU shall switch the SPDT switch to replace the MDA with the forward omni. This action guarantees that MDA pointing cannot effect command reception. At the same time the RRU shall switch the antenna/receiver transfer switch, as the new combination is more likely to be operable since it is known that the alternate configuration is likely to have a failure.

2) The RRU timer shall operate continuously. It shall be reset whenever a real time command is received, or whenever a command is received from the stored command processor to set the state of the antenna/receiver transfer switch either to its existing state or to a new state. Issuance of this latter type of command, therefore, effectively overrides RRU switching for the reset period. An override is necessary in order to minimize the switching in the rf subsystem during predictable periods when no uplink will be received by the spacecraft. Previously it was desired to override RRU operation by stopping the timer. Replacement of this override implementation eliminates a possible failure mode in time clock control circuitry and reduces pulse command requirements by two. There is also no longer a means for indefinitely disabling the RRU.

An alternate algorithm is to refrain from switching out the MDA until a certain number of hours after switching of the antenna/receiver transfer switch. This would minimize the possibility of losing the downlink when unwarranted (i.e., when an anomaly other than antenna pointing causes loss of command link). However, it is considered that antenna pointing is a more likely source of link interruption than receiver or demodulator unit failures. This algorithm is, therefore, discarded in favor of the preceding one.

Use of the polynomial code validity indicator, rather than the receiver-in-lock indicator, as a stimulus for resetting the RRU timer, results in more failure immune RRU operation. This is most apparent when only one antenna has earth visibility. If the receiver connected to this antenna is intermittently in lock; if it is falsely indicating an in-lock condition; or if it is in lock and its associated demodulator has failed such that it cannot obtain bit synchronism; then no commands can be received, yet the RRU will never alter configuration. Use of the polynomial code validity circumvents these problems. In addition, an interface between the RRU and the receivers is unnecessary. The failure mode of the polynomial code circuitry to falsely indicate valid command reception is unlikely, since the signal is in the form of pulses and most failures are of a dc nature. Should this failure occur, the alternate central decoder would be utilized for command decoding and the polynomial code check would be available from this source.

For the probe bus spacecraft,the RRU implementation will be identical. There will simply be no output to an SPDT antenna select switch since none exists.

A previous RRU functional design has been discarded. For that scheme,the RRU was basically an eight state device which controlled connections between antennas and receivers, receivers and demodulators/ central decoders. Such a technique had the advantages that receivers could receive on the same frequency and 100 percent cross-strapping was employed, giving the safest redundant system possible. The disadvantage, which led to replacement by the above RRU design, was the relatively complex implementation. It is felt that the operational difficulties imposed by frequency addressable receivers are outweighed by the simplification realized in the above design. The only cross strapping lost was in the receiver/demodulator connection.

4.2 DATA HANDLING SUBSYSTEM STUDIES

Investigations were made and tradeoff studies performed on various aspects of data handling subsystem functions in the bus/orbiter spacecraft and the large and small probes. The studies resulted in an optimum solution of system tradeoffs and in better definition of the data handling subsystem baselines. In some instances the hardware or design technique selected represents the best solution presently available and could change at a later date, based upon further definition of subsystem requirements or hardware developments.

A summary of the studies performed is given below. The principal criteria used in performing the studies and the pertinent conclusions and results are included.

1) Telemetry Data Recovery Analysis. An analysis was performed of the process of telemetry data recovery. Elements of the DSN which detect, recover, and decode telemetry data were examined. Characteristics of these elements were accounted for in order to assure data systems compatibility with the DSN and to maximize the amount of data return for both multiprobe and orbiter missions.

2) Data Handling Interface Methods. A study was performed to define a data handling interface, for both spacecraft and probes, that is optimum in terms of reliability and flexibility and that also makes maximum use of existing designs in order to minimize cost and provide a firm baseline for accurate pricing. After considering various design techniques, the data handling interface method that was selected is the same as that used on the OSO-I program. Since the OSO design was developed specifically for scientific data handling and accommodates a variety of signal interface types, it is particularly well suited for use on Pioneer Venus.

3) Orbiter Data Storage Analysis. Overall data storage requirements have been derived from data rates associated with different phases of the Pioneer Venus mission. A data storage size of 1,048,576 bits is attractive as it meets data storage requirements for the periapsis and apoapsis phases. Another tradeoff study was conducted on memory technologies for data storage use on the orbiter spacecraft. The major objectives were to determine applicability and availability for space use at minimum cost. It was determined that both magnetic core memory and semiconductor dynamic MOS memory are feasible for use on Pioneer Venus. The magnetic core memory was chosen because of its low cost, relatively low power, and the fact that it is a space proven technology and offers the minimum technical risk.

4) Probe Data Storage Hardware. A tradeoff study was also performed for data storage use in the probes. A separate study was warranted because of the considerable difference in storage requirements between the orbiter and the probes (approximately 393 kilobits for the orbiter compared to 4 kilobits for the large probe). The study determined that semiconductor memories should be used on the probes due to their small size, mass, and cost in applications requiring low storage capacity. A bipolar 256 bit memory, which is presently being space qualified, is preferred for the probe design. However, large semiconductor memories, both bipolar and MOS, are becoming available and appear to be suitable for space use. One of these may become preferable at a later time, depending upon availability and qualification costs.

5) Probe Stored Data Playback Techniques. An additional study was conducted to determine whether the stored data in the probes should be played back in a single "burst dump" of the memory or whether it should be interleaved with real time data in a single format. It was found that the burst dump technique did not allow a power reduction in the probes as originally thought, and, because of the added complexity of this technique, the approach of interleaved playback is preferred.

6) Probe Multiple Data Formats. An investigation was made to determine the impact upon the probe data handling subsystem of providing two or more downlink data formats in order to accommodate different science sampling rates during probe descent. It was found that the cost to the probe design of providing multiple data formats was minimal due to the fact that read only memory (ROM) elements are used to program the probe format. Depending upon the number of inputs and the sampling requirements of each, the cost may only entail provision of the additional command outputs necessary to change formats. It was concluded that the advantage to be gained by providing multiple formats, (i.e., to meet science data return requirements without having to add increased rf power amplication) was worth the possible increase in complexity.

7) Probe Multiple Data Rates. A study task was conducted to determine the impact upon the probes of providing two or more bit rates in the data handling subsystem, since the use of lower bit rates during the latter phases or probe descent allows a reduction of power consumption in the communications subsystem. It was determined that there is minimal cost impact if the multiple bit rates are related by an integral binary ratio (e.g., 2^n:1 such as 2:1, 4:1, etc.), or by certain nonbinary ratios (e.g., 2^n:3 such as 1:3, 2:3, 4:3, 8:3, etc.). The small additional cost is due primarily to the circuitry required to change bit rates, including the additional command outputs necessary from the command subsystem.

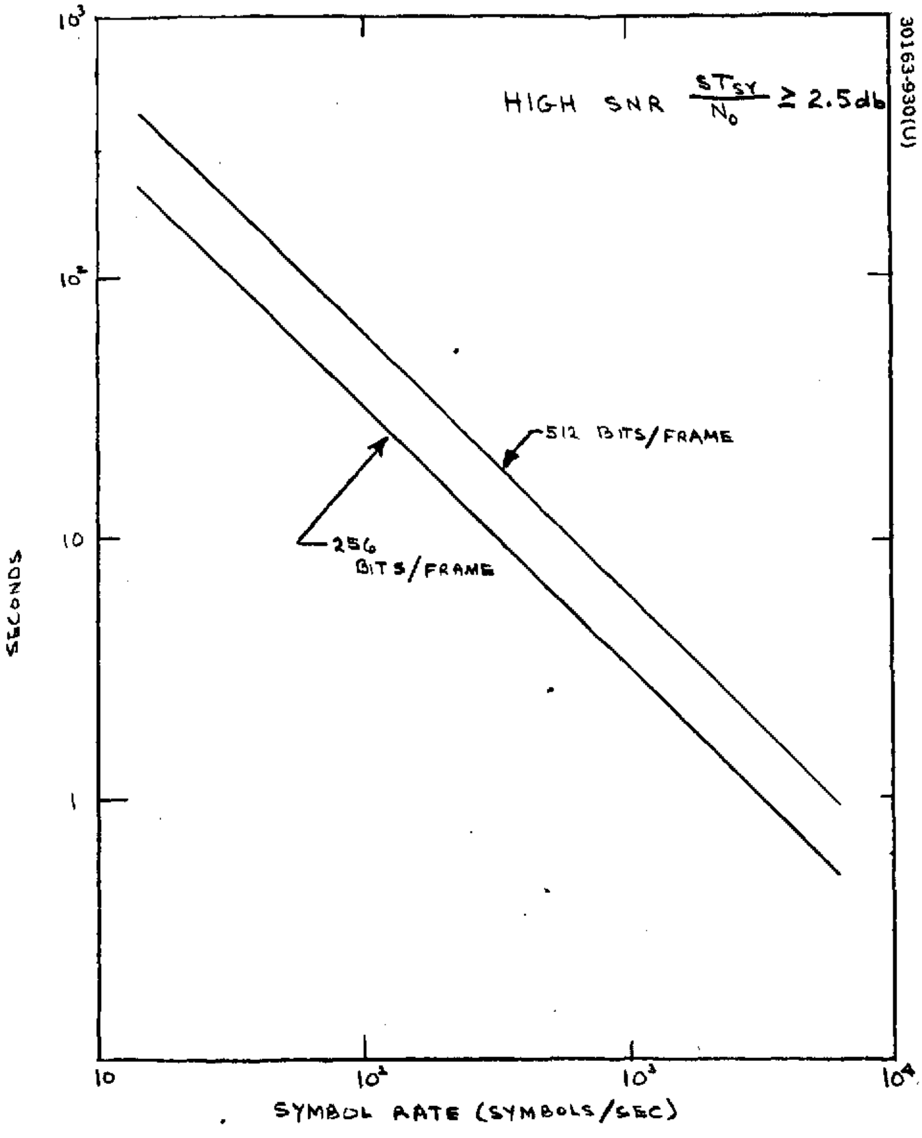


FIGURE 4-1. COMPOSITE SSA AND DDA ACQUISITION AND IN-LOCK CONFIRMATION TIME FOR SEQUENTIAL DECODING OF CONVOLUTIONAL CODED DATA (CURRENT ATLAS/CENTAUR BASELINE)

Each of these studies is described in more detail in the following sections.

Telemetry Data Recovery Analysis

Requirements for specialized hardware and software necessary to process the telemetry data to an uncoded PCM data stream have been examined. The spacecraft has been designed so that specialized equipment or software are not required. Equipment and routines presently in use on the Pioneer 10 and 11 programs are satisfactory. The probes have also been designed so that present Pioneer data processing techniques are applicable. A predetection recording scheme has been recommended for the multiprobe mission in order to maximize the probability of data capture during probe encounter and descent. It is recognized that the predetection recording and playback equipment is not presently used on the Pioneer program, since it has been used in the past on JPL Mariner programs it is not considered specialized. The following material describes the factors associated with data recovery, ground equipment limitations, and spacecraft and probe designs which maximize use of existing ground hardware and software and which minimize data loss. This discussion utilizes parameters from the current Atlas/Centaur baseline.

Spacecraft and Probe Data Characteristics

Data is obtained from the subsystems and scientific instruments and processed into a single binary PCM data stream. Analog data is converted into 8 bit digital words. Digital and discrete data are also formulated into 8 bit words.

The spectrum of the telemetry data is kept outside of the tracking loop bandwidth of the DSIF receiver by phase modulating a square-wave subcarrier with the composite PCM data. The data bit stream is modulo 2 combined with the subcarrier before phase modulating the RF carrier. Data rates and subcarrier frequencies are different for each Pioneer Venus vehicle and are summarized in Table 4-6. All are compatible with the DSN multiple mission telemetry applications (MMT) and utilize DSIF channel D, which is described in JPL document 810-5, DSN/Flight Project Interface Design Handbook.

Prior to subcarrier modulation, the data bit stream is convolutionally encoded. The convolutional encoder replaces each data bit generated with two parity bits designated P and $\overline{Q}$. The value of each parity bit is based upon the values of selected data bits previously generated in a 32 bit shift register. The code utilized is the familiar Pioneer nonsystematic quicklook code. Frame lengths for each of the Pioneer Venus vehicles coded data is shown in Figure 4-1 as a function of symbol rate for signal-to-noise ratios +2.5 dB. Curves are shown for frame lengths of 256 and 512 bits. Performance of the DDA shown in Figure 4-1 assumes that six frames are required for frame synchronization and confirmation.

TABLE 4-6. SPACECRAFT CHARACTERISTICS
(CURRENT ATLAS/CENTAUR BASELINE)

Vehicle	Bit Rates, bps	Subcarrier Frequencies, Hz	Frame Length, bits	Modulation
Probe Bus	$8/\cdots/2^N/\cdots/2048$ Non integer $3 \leq N \leq 11$	32,768	256	PCM/PSK/PM
Orbiter	$8/\cdots/2^N/\cdots/2048$ Non integer $3 \leq N \leq 11$	32,768	256	PCM/PSK/PM
Large Probe	160/80	20,480	512	PCM/PSK/PM
Small Probe	60/30/10	30,720	512	PCM/PSK/PM

While the vehicle designs are compatible with the DSN MMT system, there is the problem of the ground system being out of lock. The out-of-lock situation is particularly important to the multiprobe mission where a maximum amount of data is sought in a short time interval between encounter and destruction. The orbiter mission requires less consideration of the out-of-lock situation, due to the repetitive nature of the daily orbit and the large amount of scientific data which is stored on board the spacecraft and which can be played back repetitively as desired.

Factors that require the ground data processing system to acquire (or reacquire) lock are: 1) initial lockup, 2) channel fades, 3) frame deletions, and 4) changing bit rates and data formats.

Initial lockup is not considered a problem for the probe bus or orbiter spacecrafts. A great deal of current data will be available to allow accurate frequency predictions. Also, signal levels are sufficient to allow ready recognition of the spacecraft signal. Table 4-7 tabulates the margins for pure carrier and carrier with data for specific mission phases for all Pioneer Venus vehicles.

TABLE 4-7. MARGIN SUMMARY (CURRENT ATLAS/CENTAUR BASELINE

Vehicle	Carrier Margin*	Carrier With Data Margin
Probe Bus		
Encounter	17.1	11.5
Orbiter		
Orbit Insertion	1.4	3.6
End of Mission	2.8	0.1
Large Probe		
Acquisition after Blackout	2.7**	1.4**
20 km	2.7**	3.5**
Small Probe		
Acquisition after Blackout	0.5**	0.5**
40 km	0.6**	3.0**
20 km	2.6**	4.9**

*Margin referenced to 10.0 dB SNR in carrier loop 2 B_{LO}.

**Includes 2 dB predetection recording loss.

Initial lockup for the large and small probes could require more time due to carrier frequency dispersions at encounter. These dispersions are the result of the combination of long term drift, doppler uncertainty, and thermal effects. For this reason: 1) the JPL scheme for predetection recording has been recommended, and 2) the stored sequences for both the large and small probes provide a 15 min period of unmodulated carrier prior to the blackout phase. Predetection recording eliminates the requirement for receiver lockup. The unmodulated carrier provides a strong signal for a period of time that could be used for a check of the ground equipment setup.

TABLE 4-8. CONVOLUTIONAL CODED TELEMETRY CHARACTERISTICS (CURRENT ATLAS/CENTAUR BASELINE)

Characteristic	Probe Bus and Orbiter	Large Probe	Small Probe	DSIF Capability
Constraint length, bits	32	32	32	32 (maximum)
Frame length, bits	256	512	512	1200 (maximum)
Coder connection vector	Nonsystematic quick-look	Nonsystematic quick-look	Nonsystematic quick-look	Nonsystematic or systematic
Code rate	1/2	1/2	1/2	1/2
Bit rate, bps	8/.../2^N/.../ 2048	160/80	60/30/10	2048 (maximum) 6 (minimum)
Tail length, bits	24	24	24	8 to 48

The large and small probe channel fades are caused by turbulence in the Venus atmosphere. An allowance for fades (as derived from the Stanford analysis of the Venus atmosphere stochastic effects) has been included in the telecommunications link performance.

Frame deletions are caused by buffer overflow in the DDA during the sequential decoding process. Discussions of the Fano algorithm and the associated decoding factors are available in the literature. Table 4-8 lists the telemetry characteristics and DSIF capability. The buffer overflow problem is reduced by short telemetry frame lengths and low bit rates. In addition, the buffer overflow problem on the multiprobe mission is further reduced as the predetection recording can be played back at slower than real time rates to allow increased computation time per information bit.

Figures 4-2 and 4-3 portray the data which would be lost due to changing data rates in typical large and small probe missions. The loss of data is attributable to loss of SSA and DDA synchronization and is equal to approximately 3000 bits in every case. Several schemes have been examined to minimize or eliminate this source of data loss. These are discussed in a separate trade study. The most attractive scheme at present is to provide a separate data buffer which would, in parallel with the real time link, store data during the reacquisition sequence. This stored data would be interleaved in the telemetry format for replay subsequent to reacquisition. The buffer would be utilized during initial acquisition and at every subsequent bit rate change. The bit rate changes are required to maximize the data which will be returned for a given RF power output level.

At probe bus encounter, the telemetry symbol rate is 4096 SPS and the reacquisition process requires less than 1 sec.

Telemetry formats for all vehicles have been designed so that format changes do not result in loss of ground system lock. All frame sync words and tail sequences are identical for all formats.

Data Handling Interface Methods

The purpose of this study was to define an optimum data handling interface method for gathering data from science instruments and engineering equipment on the four Pioneer Venus space vehicles. The particular interface to be investigated was that between the data handling subsystems and the users. A user is defined as the source of the data to be gathered, which may be a science instrument or another engineering subsystem. The study was to include all analog and digital data requiring sampling by the data handling subsystem; the types of digital data to be considered included both serial digital signals and bilevel discrete signals.

The fundamental objectives and the procedural approach for this study were the same as those described for the study of command interface methods in subsection 4.1. First, interface guidelines were established

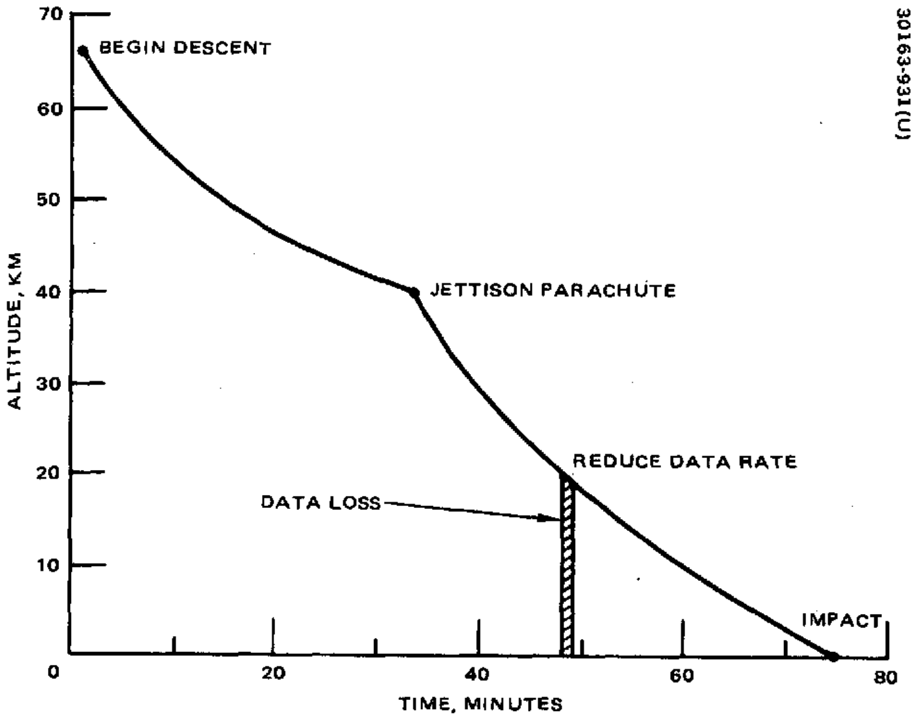


FIGURE 4-2. LARGE PROBE DESCENT PROFILE (CURRENT ATLAS/ CENTAUR BASELINE)

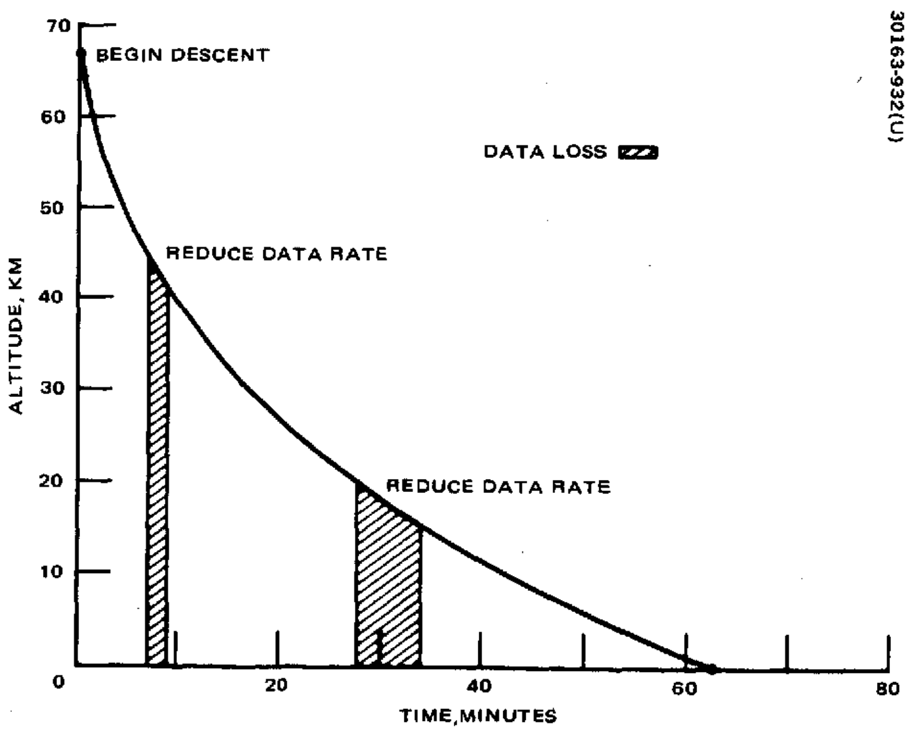


FIGURE 4-3. SMALL PROBE DESCENT PROFILE (CURRENT ATLAS/ CENTAUR BASELINE)

which defined optimum data handling interface features and characteristics for scientific type spacecraft. A major guideline was that the interface must be sufficiently flexible to handle a multiplicity of science instrument and equipment complements with maximum modularity and standardization; also, particular emphasis was placed on grounding techniques that provide isolation of signal reference grounds so as to enhance data point measurement accuracy. Then, a survey was conducted to solicit information on previously developed data handling subsystem hardware from known suppliers of space equipment. Details of this survey are reported in subsection 4.3. Using information from the survey, an evaluation of existing equipment designs from six different companies resulted in selection of the data handling interface hardware developed for the OSO-I program.

The OSO interface hardware is felt to be an optimum choice for all four space vehicles. After the surveyed equipment was evaluated, it was found that most of the other equipment designs met portions of the study objectives and guidelines to varying degrees. However, the OSO hardware was the only interface equipment located that meets all of them without modification. The OSO interface hardware consists of standard data input modules; these modules are called remote multiplexers in OSO terminology. The modules are a modern design and employ custom LSI technology to reduce mass and improve reliability. They were developed specifically for use on scientific spacecraft. The design of these modules is particularly well suited for Pioneer Venus since the interface requirements for these four spacecraft are very similar to those for OSO-I.

The following subsections present a summary description of the selected interface design as applied to the Pioneer Venus data handling subsystems; a more detailed description may be found in Reference 4-2. First, a general description of the design aspects common to both the bus/orbiter spacecraft and to the probes is described. Then, the unique characteristics of the interface design for each vehicle are discussed separately.

General Interface Description

The data handling subsystem accepts both analog and digital input data. In addition, it provides output timing signals to user subsystems or science instruments in order that they may transfer serial data or perform other operations in synchronization with the data handling subsystem. There is considerable commonality between input and output circuitry of the bus/orbiter spacecraft and the probes; thus, the input data interface and the output timing interface are quite similar for all vehicles.

Three types of signals are accepted by the data handling subsystems of the bus/orbiter spacecraft and the probes: analog, serial digital, and bilevel discrete signals. The input multiplexer of the data handling subsystem will utilize the same circuitry for all three signal types in both the spacecraft and the probes; thus, much of the input specification is the same for all subsystems and science instruments on all vehicles. Certain specifications, such as sampling time, will vary with a particular vehicle, its mode of operation, and the input signal type. A complete definition of these parameters will be made during final system design.

The data handling subsystem input specifications are summarized in Table 4-9.

TABLE 4-9. DATA HANDLING SUBSYSTEM INPUT SPECIFICATION

Multiplexer input impedance	
Input current	
Multiplexer ON, sampling time	± 1.0 μA maximum for $-1\text{ V} \le V_{in} < +5.3$ V; +300 μA maximum for $+5.3\text{ V} \le V_{in} < +17$ V
Multiplexer ON, nonsampling time	± 0.1 μA maximum for $-1\text{ V} \le V_{in} < +5.3$ V; +300 μA maximum for $+5.3\text{ V} \le V_{in} < +17$ V
Multiplexer OFF	± 0.1 μA maximum for $-1\text{ V} \le V_{in} < +12$ V; +300 μA maximum for $+12 \le V_{in} < +17$ V
Input capacitance	300 pF maximum (plus up to 700 pF line capacitance).
Multiplexer sample time	Variable, depending upon operating mode (TBD). See text.
Data source failure modes	
Open circuit	Undefined output
Short circuit to low impedance source	-50 to +50 V with no damage; other channels are not affected. For voltage outside this range, special precautions must be taken during the design phase to protect the multiplexer.

Data source requirements differ with the signal type and, for analog signals, with the A/D conversion accuracy required. These data source output requirements are summarized in Table 4-10. Individual signal types are discussed below.

The data handling subsystem samples analog inputs and converts them into serial digital words for telemetering to the ground. In the bus/orbiter spacecraft, analog signals are converted into 8 bit words; in the probes, they are converted into 10 bit words since the accuracy requirement is greater for certain specified inputs.

The term analog as used here covers two different types of measurements. One is a dc voltage with a range of 0 to +5.12 V, which would typically be the output of a buffer amplifier in a vehicle subsystem or science instrument. As measured, the other is also a 0 to +5.12 Vdc signal, but it is generated by supplying a constant current (1 mA) to a variable resistor (thermistor, potentiometer, etc.) in a vehicle subsystem or science instrument. The current is generated in the telemetry subsystem and is gated to the variable resistor during the sampling interval on the multiplexer input line assigned to that particular analog (resistance) measurement. The variable resistor must operate over a range of 0 to 5.12 kΩ.

Each multiplexer can accept a signal return line, from a given using subsystem or science instrument, for differential measurement of analog data signals. This line is terminated with a high impedance inside the data handling subsystem, ensuring essentially no current flow, and is used as an analog reference return.

Discrete bilevel signals to be telemetered are sampled and assembled by the spacecraft (probe) multiplexers into 8 (10) bit words for transmission. In addition to accepting discrete bilevel signals, bulk scientific data can also be accepted in 8 (10) bit bytes on 8 (10) parallel lines during the time interval defined by a read envelope. Multiple 8 (10) bit bytes can share these 8 (10) lines, resulting in the data being located at the desired positions throughout a minor frame if the falling or leading edge of the read envelope is used to update an 8 (10) bit output register within the science instrument. Minor frame sync is provided by the data handling subsystem to enable the science instrument to synchronize the gathering of experiment data, and thus to distinguish one 8 (10) bit byte from another.

For telemetering large quantities of data from an individual spacecraft subsystem or science instrument, a serial digital data interface is preferred as compared with the use of a parallel data transfer via multiple bilevel digital inputs; this is because the serial approach reduces multiplexer hardware and simplifies control logic. However, analog and discrete bilevel inputs are also provided for use as required.

TABLE 4-10. DATA SOURCE OUTPUT REQUIREMENTS

Analog data	
Data voltage range	0 to +5.120 V
Source impedance	10 kΩ maximum, 8 bit accuracy (100 ohms maximum, 10 bit accuracy, probes only)
Grounding	Users shall supply their local signal ground to the multiplexer as a reference. This line is terminated with a high impedance inside the data handling subsystem. Inputs will be handled differentially with respect to this signal ground.
Bilevel discrete data	
Logical 0	-1 to +1 V
Logical 1	+4 to +17 V
Source impedance	10 kΩ maximum
Serial digital data	
Logical 0	-1 to +1 V
Logical 1	+4 to +17 V
Voltage rise and fall times (to or beyond logic threshold)	3 μsec maximum with 1000 pF capacitive load (300 pF maximum multiplexer capacitance plus 700 pF maximum line capacitance (≤14 ft of cable)
Delay time	Data bit 1 shall be at or beyond logic threshold within 40 μsec after user's read envelope input buffer's collector output switches to logic 0 (logic "0" to "1" transition at input). Data bits 2 through 8 (2 through 10 in probes) shall be valid within 6 μsec after user's read clock buffer's collector output switches to logic 1 (logic "1" to "0" transition at input).
Data sync	Data transitions shall occur with the 1 to 0 transition (trailing edge) of clock pulses

NRZ serial digital data sources are provided with a read envelope and a read clock signal in synchronism with which an 8 (10) bit word group is serially shifted to the multiplexers. Each serial user should enable the common read clock signal, with the unique read envelope signal provided to him for data readout at the correct times. Internal submultiplexing can be performed by the user in 8 (10) bit bytes to time share one data channel if the data readout is properly synchronized with the data handling timing signals (e.g., minor frame or major frame sync) so that the data can be identified in ground processing. Alternately, multiple 8 (10) bit groups can share a common data bus output (i.e., wire ORing the data channels together) through proper steering by unique read envelope signals, one for each 8 (10) bit group.

The data handling subsystem provides read envelope signals to users for gating of their bilevel discrete outputs; it provides both read envelope and read clock signals to users for gating of serial digital outputs. In addition, various clock frequencies are provided for general use to vehicle subsystems or science instruments. Output specifications for the data handling subsystem are summarized in Table 4-11.

Read envelope signals are provided to bilevel data sources and to serial data sources. Bilevel data sources may use this signal for synchronization or, in conjunction with frame sync signals, for time sharing output data lines within the user subsystem or instrument. Serial data sources may also use the read envelope signal in this manner; however, it also defines the time during which serial data must be clocked out to the data handling subsystem by the read clock signal.

Read clock signals are provided to all serial data sources and are used to serially shift data to the data handling subsystem.

In general, the frequency and pulsewidth of the read clock and read envelope signals will vary with the particular vehicle and also with the operating mode if multiple bit rates are used; complete definition will be made during final system design.

The data handling subsystem provides major frame sync, minor frame sync, and word sync signals such that submultiplexing of data can be accomplished within a user subsystem or science instrument. These signals can also be used to notify a user that he will be sampled, at a fixed later time, in order to provide him time to get his data ready. Only square wave frequency sources that can be derived from a simple countdown chain will be provided. The key output frequencies will have a period equal to the major frame period, the minor frame period, the word period, and the bit period. If needed, other frequencies can easily be derived within a science instrument by using one of these signals, which represents the minimum resolution desired, to drive a countdown chain within the science instrument; another one of these signals (such as major frame sync) can be used to reset this counter at the appropriate time in order to maintain a proper phase relationship with the clock. In using this two signal interface, one could greatly reduce the number of interface wires and interface buffers between the data handling subsystem and all of the using subsystems and instruments. On the other hand, the complete countdown does exist within the data handling subsystem and any of its outputs could be provided to an instrument if needed.

In addition to the above frequencies which, in general, will vary with changes in the bit rate or data format, at least one fixed frequency (e.g., 2048 Hz) will be provided by the data handling subsystem.

Where multiple bit rates are employed (i.e., where bit rates are changed for different operating modes), "bit rate mode" signals will be made available to users to indicate when a particular bit rate is in use. These signals can be used to inhibit selected frequencies when a user subsystem is not being sampled by the data handling subsystem.

Unique Spacecraft Characteristics

The data handling subsystem of the bus/orbiter spacecraft provides the means for retrieval of spacecraft status and experimental data. The subsystem provides full redundancy in all mission critical areas and achieves high reliability. Redundant signals will be connected to two different remote multiplexers. In case of a single multiplexer failure, the redundant line will still be sampled.

All remote multiplexer units are identical and interchangeable. Each unit may be separately programmed to operate in one of four different modes by external wiring of four unit connector pins. For mode definition, see Table 4-12. Table 4-13 presents this same mode data in a different format.

The four modes define which of the 32 input channels will be used for the three different types of input data. Table 4-13 indicates the type and quantity of each input in each of the four modes. In the mode 1 configuration, all 32 inputs are bilevel digital data. In the mode 2 configuration, there are 24 bilevel digital inputs and five inputs which may be either analog or serial digital. In the mode 3 configuration, there are two different input choices; these are essentially two separate submodes that require specific programming of the fixed memory in the telemetry processor. In one of these submodes, there are 16 bilevel inputs and 14 inputs which may be analog or serial digital. In the mode 4 configuration, there are 16 analog inputs and 16 inputs which may be analog or serial digital.

It may be noted from the tables that the same inputs may be used for analog, bilevel, or serial data, depending on the mode selection. Therefore, several multiplexer characteristics are common to any type of input as previously summarized in Table 4-9.

TABLE 4-11. DATA HANDLING SUBSYSTEM OUTPUT SIGNAL SPECIFICATION

Read envelope signal	
Waveform	Rectangular
Pulsewidth Frequency	Variable, depending upon operating mode (TBD). See text.
Fanout capability	Two (with standard input buffer, Hughes Part No. 908974)
Logical 0	0 V (referenced to the data handling subsystem signal return) through 5.3 ±2.9 kΩ source impedance
Logical 1	+13 ±3 volts capable of supplying 4 mA.
Voltage rise time (10 to 90 percent)	2.5 μsec maximum. Load equals 700 pF line capacity in parallel with a standard input buffer (or 30 kΩ). This corresponds to a total line length ≤ 14 ft.
Voltage fall time (90 to 10 percent)	15 μsec maximum. Load equals 700 pF line capacity in parallel with a standard input buffer, Hughes Part No. 908974.
Short circuit protection	Current limiting is provided for protection against shorts to ground
Read clock signal	
Waveform	Rectangular, gated
Frequency	Variable, depending upon operating mode (TBD). See text.
Fanout capability	Twelve (with standard input buffer, Hughes Part No. 908974)
Logical 0	0 V (referenced to the data handling subsystem signal return) to +1 V while sinking 100 μA maximum
Logical 1	+13 ±3 volts capable of supplying 24 mA
Voltage rise time (10 to 90 percent)	2.5 μsec maximum. Load capacitance shall be 3000 pF maximum

Table 4-11 (concluded)

Voltage fall time (90 to 10 percent)	5.0 μsec maximum. Load capacitance shall be 3000 pF maximum.
Number of clock pulses	Eight (ten in probes) per read envelope
System clock signals	
Waveform	Square wave with 50 percent ±2.5 μs duty cycle
Frequency	Multiple and variable. See text.
Fanout capability	Two (with standard input buffer, Hughes Part No. 908974
Logical 0	0 to +1 V
Logical 1	+14 ±2 V capable of supplying 4 mA
Voltage rise and fall times	2.5 μsec maximum with 3000 pF maximum load capacity
Short circuit protection	Current limiting is provided for protection against shorts to ground
Bit rate mode signals	
Waveform	Logical 0 or logical 1, depending upon operating mode.
Fanout capability	
Logical 0	
Logical 1	
Voltage rise time	Same as read envelope signal
Voltage fall time	
Short circuit protection	

TABLE 4-12. TELEMETRY REMOTE MULTIPLEXER INPUT OPTIONS

Input Address MSB LSB	Read Envelope Selection	Read Clock Present	Mode 1			Mode 2			Mode 3			Mode 4		
			Data Type[1]	Sampled Input Channel(s)	Applicable Current Bus[3]	Data Type	Sampled Input Channel(s)[2]	Applicable Current Bus[3]	Data Type	Applicable Input Channel(s)[2]	Applicable Current Bus[3]	Data Type	Sampled Input Channel(s)	Applicable Current Bus[3]
00000	0	Yes	P	24-31	-	P	24-31	-	P	24-31	-	A or S	0	1
00001	1	Yes	P	16-23	-	P	16-23	-	P[4]	16-23	-	A or S	1	1
00010	2	Yes	P	8-15	-	P	8-15	-	A or S	2	1	A or S	2	1
00011	3	Yes	P	0-7	-	A or S	3	1	A or S	3	1	A or S	3	1
00100	4	Yes				A or S	4	2	A or S	4	2	A or S	4	2
00101	5	Yes				A or S	5	2	A or S	5	2	A or S	5	2
00110	6	Yes				A or S	6	2	A or S	6	2	A or S	6	2
00111	7	Yes				A or S	7	2	A or S	7	2	A or S	7	2
01000	8	Yes							A or S	8	3	A or S	8	3
01001	9	Yes							A or S	9	3	A or S	9	3
01010	10	Yes							A or S	10	3	A or S	10	3
01011	11	Yes							A or S	11	3	A or S	11	3
01100	12	Yes							A or S	12	3	A or S	12	3
01101	13	Yes							A or S	13	3	A or S	13	3
01110	14	Yes							A or S	14	3	A or S	14	3
01111	15	Yes							A or S	15	3	A or S	15	3
10000	--	No							A	16	4	A	16	4
10001	--	No							A	17	4	A	17	4
10010	--	No							A	18	4	A	18	4
10011	--	No							A	19	4	A	19	4
10100	--	No							A [4]	20	5	A	20	5
10101	--	No							A	21	5	A	21	5
10110	---	No							A	22	5	A	22	5
10111	--	No							A	23	5	A	23	5
11000	--	No										A	24	6
11001	--	No										A	25	6
11010	--	No										A	26	6
11011	--	No										A	27	6
11100	--	No										A	28	6
11101	--	No										A	29	6
11110	--	No										A	30	6
11111	--	No										A	31	6
Mode pin Connection			A to return; B to return			A to return; B = open circuit			A = open circuit; B to return			A = open circuit; B = open circuit		

[1] "P" is parallel bilevel digital, "S" is serial digital, "A" is analog.

[2] Input channels 0, 1, and 2 are not used in mode 2 and channels 0 and 1 are not used in mode 3.

[3] Use of a current bus for a signal conditioned channel commits the other channels associated with that bus to also be used for signal conditioning.

[4] In mode 3 input channels 16 to 23 may be used for parallel bilevel data (address 00001) or analog data (addresses 10000-10111).

TABLE 4-13. TELEMETRY REMOTE MULTIPLEXER INPUT OPTIONS

Serial Inputs (8-bit bytes)	Analog Inputs (0 to +5.12 volts)	Parallel Bilevel Words (8-bit bytes)	Mux Mode	Number of Input Lines Used
16	16	0	4	32
15	17	0	4	32
14	18	0	4	32
14	8	1	3	30
14	0	2	3	30
13	19	0	4	32
13	9	1	3	30
13	1	2	3	30
12	20	0	4	32
12	10	1	3	30
12	2	2	3	30
11	21	0	4	32
11	11	1	3	30
11	3	2	3	30
10	22	0	4	32
10	12	1	3	30
10	4	2	3	30
9	23	0	4	32
9	13	1	3	30
9	5	2	3	30
8	24	0	4	32
8	14	1	3	30
8	6	2	3	30
7	25	0	4	32
7	15	1	3	30
7	7	2	3	30
6	26	0	4	32
6	16	1	3	30
6	8	2	3	30
5	27	0	4	32
5	17	1	3	30
5	9	2	3	30
5	0	3	2	29
4	28	0	4	32
4	18	1	3	30
4	10	2	3	30
4	1	3	2	29
3	29	0	4	32
3	19	1	3	30
3	11	2	3	30
3	2	3	2	29
2	30	0	4	32
2	20	1	3	30
2	12	2	3	30
2	3	3	2	29
1	31	0	4	32
1	21	1	3	30
1	13	2	3	30
1	4	3	2	29
0	32	0	4	32
0	22	1	3	30
0	14	2	3	30
0	5	3	2	29
0	0	4	1	32

The signal input capability of the telemetry subsystem is flexible. It can be changed by the addition or deletion of remote multiplexers to accommodate requirements for either more or less signal inputs.

When redundancy is required, the signals must be connected to two different multiplexers. Cross-strapping at this interface is not normally necessary since there is complete cross-strapping between the remote multiplexer and the PCM encoder. For cases where cross-strapping of multiplexer inputs may be advantageous, a description of the recommended techniques for analog, serial digital and parallel digital (discrete) data inputs may be found in Reference 4-2.

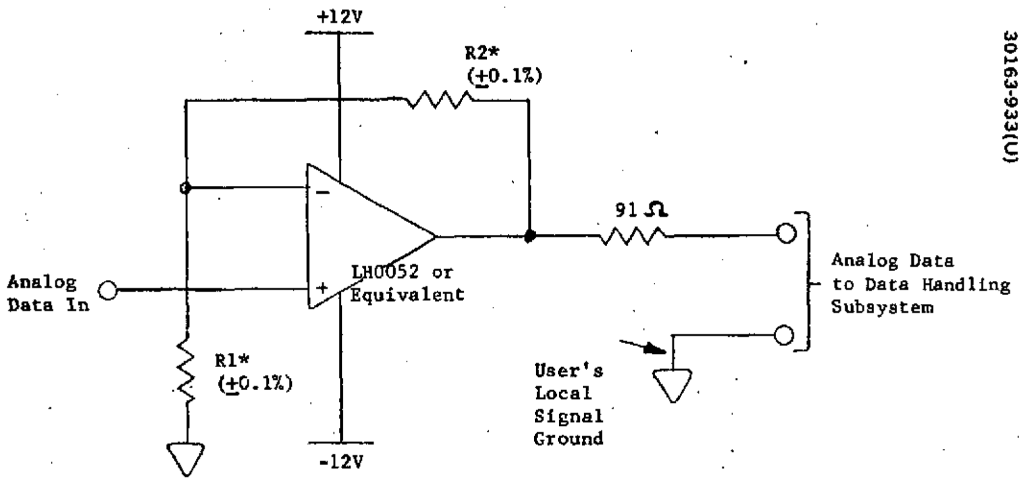
Unique Probe Data Handling Characteristics

The data handling subsystem of the probes differs from that of the bus/orbiter spacecraft in two principal respects; first, because of the size, the multiplexing is centralized rather than distributed among remote multiplexers; and second, as specified, the probe subsystem has the capability for converting analog input signals into 10 bit digital words with an accuracy of ±1.0 percent. The latter results in an analog data interface definition different from that of the spacecraft for those inputs requiring high accuracy conversion.

The data handling multiplexer has 112 input channels in the large probe and 52 input channels in the small probe. The distribution of analog, bilevel digital, and serial digital channels is given in Table 4-14. The multiplexer provides special circuitry for analog inputs requiring high accuracy A/D conversion. The A/D converter in the data handling subsystem

TABLE 4-14. PROBE DATA HANDLING MULTIPLEXER INPUT PROVISION

Input Type	Provision for Inputs	
	Large Probe	Small Probe
Analog		
10 bit accuracy	32	8
8 bit accuracy/10 bit resolution	44	20
Bilevel digital	28	20
Serial digital	8	4
Totals	112	52



* R1, R2 Selected for 0.000 to 5.120 Volt Output Range

FIGURE 4-4. HIGH ACCURACY ANALOG DATA OUTPUT CIRCUIT RECOMMENDATIONS converts the sampled analog data into a 10 bit serial word format. The bilevel digital data are formatted into 10 bit digital words, as are the serial digital data inputs, in a controlled time sequence. All three types are combined, encoded, and transmitted as a sequence of 10 bit serial data words to both the probe bus and the communications subsystem.

Analog signals to be telemetered are sampled by the multiplexer to generate a 10 bit telemetry word for transmission. Within the full-scale A/D conversion range, certain analog input signals will be converted with an accuracy of ±5 mV or ±0.1 percent of full scale (i.e., ±2.5 mV offset and ±2.5 mV quantization error). All other analog inputs will be converted with 10 bit resolution, but with ±20 mV or ±0.4 percent of full scale accuracy (i.e., ±17.5 mV offset and ±2.5 mV quantization error).

The data source requirements for high accuracy inputs are as follows:

1)	Data voltage range	0 to +5.120 V
2)	Source impedance	100 ohms maximum (see recommended circuit, Figure 4-4)
3)	Grounding	Users shall supply their local signal ground to the multiplexer as a reference. This line is terminated with a high impedance inside the data handling section. High accuracy inputs will be handled differentially with respect to this signal ground.

Orbiter Data Storage Analysis

Data storage is required on the orbiter spacecraft. Overall requirements for the data storage units have been derived from data ratio associated with different phases of the Pioneer Venus mission. Trade studies were then conducted on memory technologies for data storage use.

Requirements Analysis

Overall requirements for the data storage unit have been developed from an examination of science data requirements and the mission operation. The material presented is based upon the current Atlas/Centaur baseline:

1) The maximum storage data rate is a function of the science data requirement. This maximum rate always occurs around the periapsis point during an occulted periapsis pass. The present payload rate requirement is 490 bps, which is exclusive of overhead for sync words, engineering data, format inefficiencies, etc. The derived rate requirement (discussed later) is 640 bps which includes overhead factors.

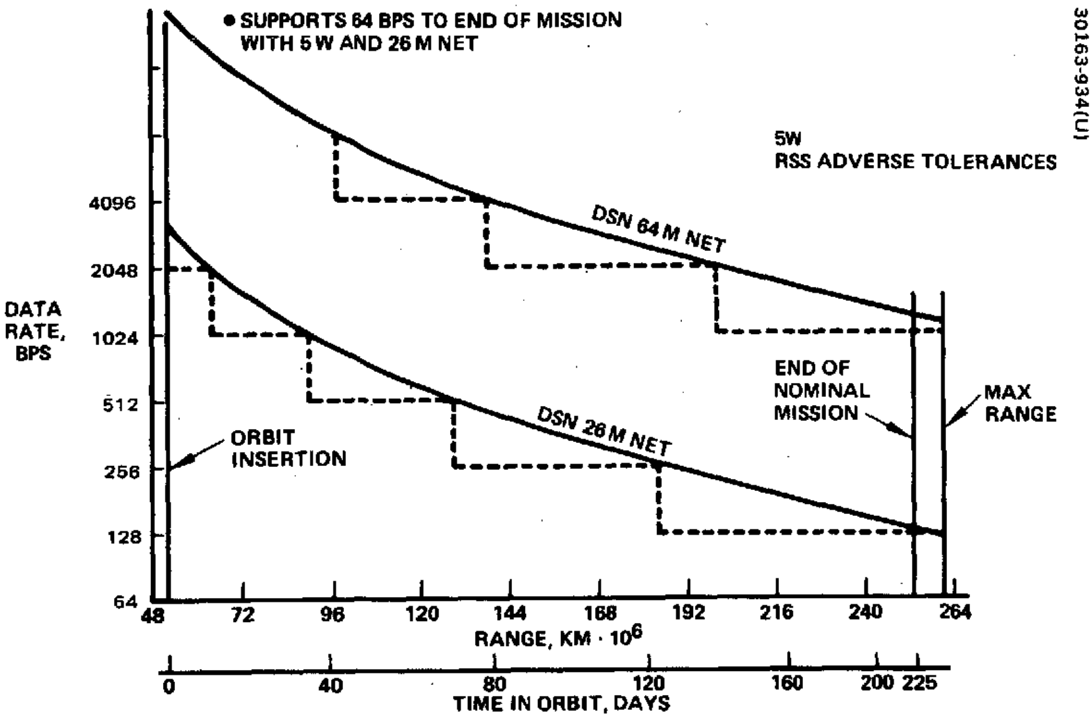


FIGURE 4-5. TELECOMMUNICATIONS PERFORMANCE DURING ORBIT PHASE OF MISSION

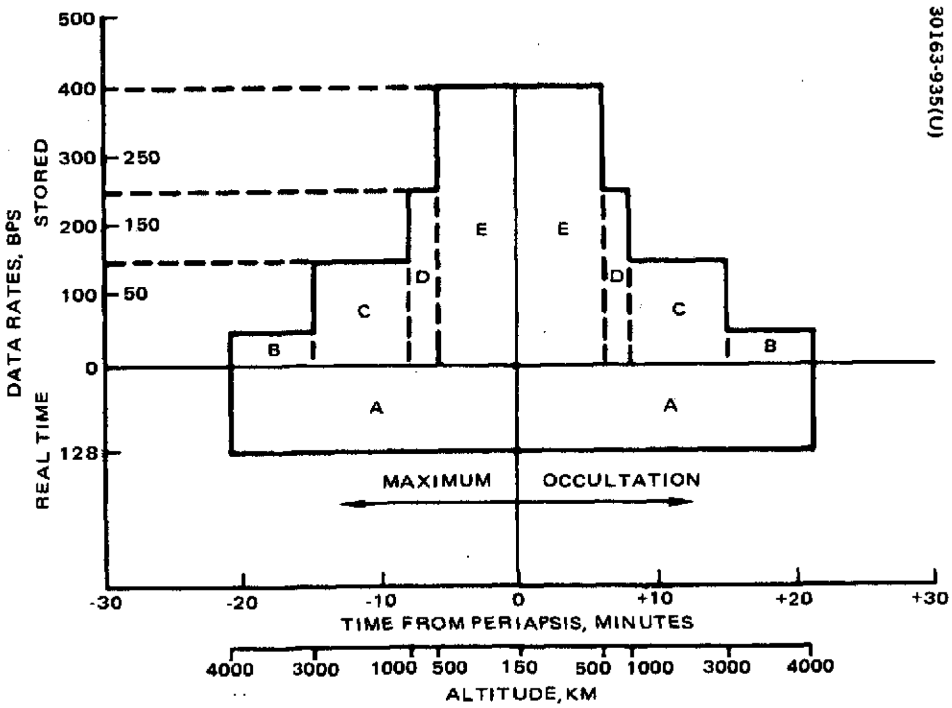


FIGURE 4-6. MINIMUM SCIENCE STORAGE REQUIREMENTS

2) The minimum data storage rate occurs after the apoapsis phase. The present science data rate requirement for this phase is 7.67 bps which is again exclusive of overhead, etc.

3) The minimum/maximum playback (or dump) rates are constrained by the downlink design which is capable of rates between 8 and 2048 bps in multiples of 2^n.

4) The minimum data storage capacity requirement is determined by data requirements during the occulted periapsis passes. The derived requirement (discussed later) is 890,880 bits.

5) The data storage unit can be operated in a simplex mode in that the capability to concurrently read and write is not required.

6) Graceful degradation is provided by supplying the unit in two separately operated modules, each of which provides half of the total required storage capacity.

As a result of the hardware study which follows, magnetic core storage has been shown to be the most acceptable implementation. Due to core memory addressing register technology, data storage capacity generally is designed in multiples of 2^n times the storage word size. The Pioneer Venus telemetry word size is 8 bits and the minor frame length is 256 bits; so the storage word size should be an integer (N) times 8 where $M(Nx8) = 256$ and M is also an integer.

As a result of countdown circuitry design considerations, data rates are also in 2^n submultiples.

To minimize circuitry onboard the spacecraft, it is desirable to minimize the total number of different data formats which would be required to store or transmit in real time.

Figure 4-5 portrays the downlink telecommunications performance for the entire orbit phase. The data rate for the first 35 days in orbit is 1024 bps, and for the next 40 days it is 512 bps, both of which meet or exceed real time science data requirements. When the link capability is reduced to 256 and finally 128 bps, the science requirements exceed the real time transmission capability and the excess is stored for later transmission. Earth occultation periods and eclipses must also be accounted for with storage and their requirements are also analyzed.

Figure 4-6 depicts the minimum science data storage requirement for an orbit late in the mission when the downlink capability is 128 bps. As shown, five data formats are utilized to provide experiment sampling. In addition, four data rates are required, none of which are binary multiples of each other.

TABLE 4-15. PERIAPSIS SCIENCE DATA FORMATS

Format	Data Rate, bps	Altitude, km	Utilization, min		Mag	Solar Wind	Electron Temperature	Neutral Mass Spec	Ion Mass Spec	UV Spec	IR Radiation	Radar Altitude
			Real Time	Stored								
A	128	<4000	42	24	32	–	24	–	–	34	–	–
					33.7*		26.2*			37.5*		
B	64	<4000	12	12	–	–	–	25	25	–	–	–
								26.2*	26.2*			
C	170-2/3	<3000	14	14	–	–	–	25	25	–	100	–
								25*	25*		100*	
D	341-1/3	<1000	4	4	–	–	–	25	25	–	100	100
								30*	30*		110*	110*
E	512	<500	12	12	–	–	–	100	100	–	100	100
								105*	105*		105*	105*

*Implemented sample rates includes 16 percent overhead provision for frame sync and engineering data transmission.

The storage requirement for this minimum implementation is 510,000 bits. This provides storage only for the science instrument data, with no provision for frame sync or engineering data. In addition, the bit rates require extra countdown circuitry as they are not binary submultiples.

Alternate format characteristics are tabulated in Table 4-15. The first format "A" is sampled at 128 bps, and the science instrument sampling requirements along with the implemented provisions are shown. For all formats, the implementation exceeds the science requirement. Data rates of 170-2/3 and 341-1/3 for formats "C" and "D" were selected because they are integer fractions of 512 bps (1/3 and 2/3's, respectively). The data sampling rate includes a minimum 16 percent overhead allowance for transmission of frame sync and engineering data. The implementation penalty associated with the overhead and "nice" data ratesis 129,936 bits (or a total storage requirement of 510,000 + 129,936 = 639,936).

An additional alternative would be to implement only binary bit rates. The results of this and the previous alternative are shown by cross hatching on Figure 4-7. This alternative, though costing 112,956 bits (total 752,892), is attractive as extra bit rate countdown circuitry can be eliminated and the entire format "D" can be deleted.

The maximum occultation period is 23 minutes. It occurs near, but is not centered on, periapsis. For computation of storage requirements, occultation is assumed periapsis centered as it represents the worst case. From Figure 4-7 it can be seen that data will be stored for part of formats "A" and "C" and all of format "D/E." The worst case number of bits required to be stored are 890,880.

Alternate configurations would minimize data storage required at the expense of additional data rates and formats. For example, if format "D" was utilized and two additional bit rates (Figure 4-7) were implemented (A + C and A + D), approximately 174,080 bits could be saved and the occulted periapsis storage would be 716,800 bits.

The maximum storage rate is reached when format "E" is sampled at 512 bps and format "A" is sampled at 128 bps. The sum is 640 bps, which is 5/8 of 1024.

An additional use of the data storage could occur at the apoapsis phase. Data storage at this phase could accommodate planned DSN outages, etc. The apoapsis phase is defined as the apoapsis point ±8 h. The science data requirement is 7.67 bps during apoapsis. This requirement is met with a spacecraft standard 16 bps clock which allows sample margin (50 percent) for sync and engineering data. At 16 bps, 57,600 bits would be stored per hour, or a total of 921,600 bits would be stored for 16 h.

An alternative configuration would utilize a nonbinary bit rate of 10-2/3 bps (which allows ample overhead provision), or for 16 hours would require 614,400 bits of storage.

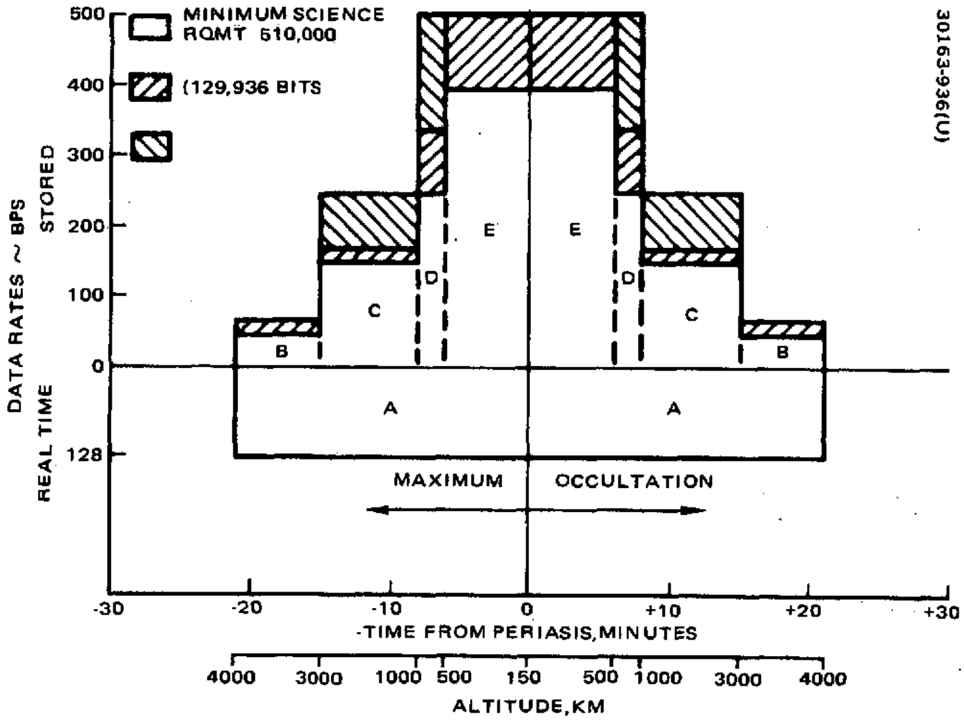


FIGURE 4-7. PERIAPSIS DATA REQUIREMENTS

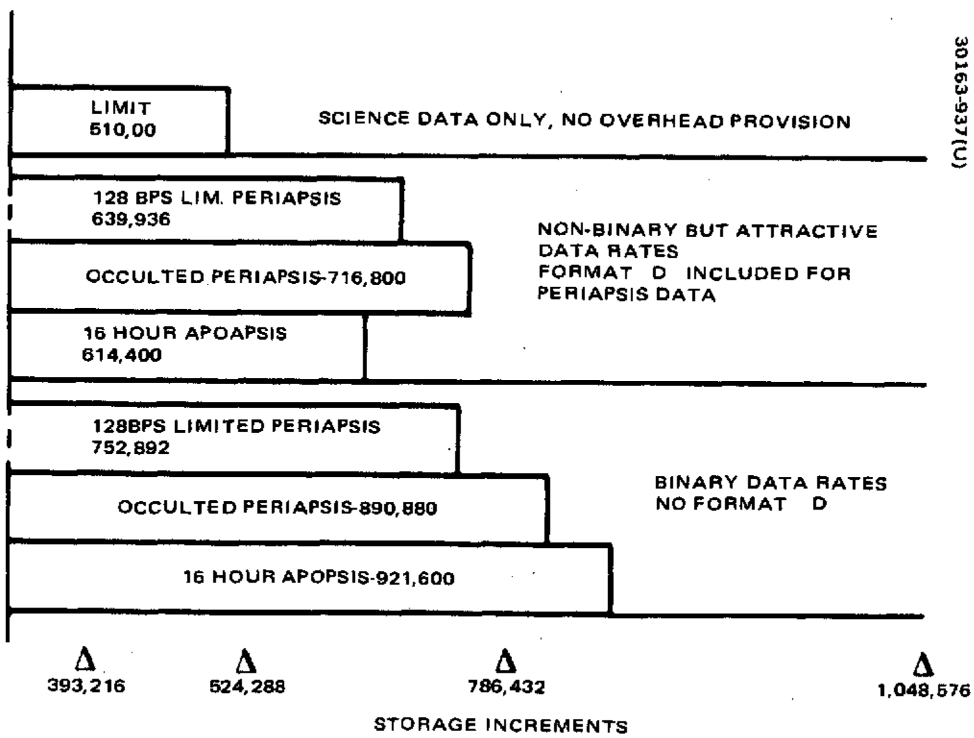


FIGURE 4-8. STORAGE CAPACITY TRADE

Typical storage capacities are shown in Table 4-16. In the region of interest, selectable capacities are 524,288; 589,824; 786,432; or 1,048,576 bits. Since it is desirable to implement the design in two independent half-capacity modules in order to provide graceful degradation in the event of failure, the 589,824 configuration can be eliminated.

Figure 4-8 relates the various storage capacities to the range of storage requirements. It can be seen that, in terms of spacecraft design minimization, the 1,048,576 configuration is optimum as it meets storage requirements for all mission phases without additional data rates (except for occulted periapsis 640 bps) or formats. The two half-capacity modules would contain 524,288 bits.

A viable alternative to the 1,048,576 configuration is 786,432. Spacecraft cost for this alternative consists of, at worst case, the inclusion of nonbinary data rates and an additional data format to handle the occulted periapsis phase. Depending upon the length of time apoapsis data is to be stored, a nonbinary rate may be desirable anyway for this phase.

Table 4-17 relates data storage implementation to spacecraft requirements for bit rates and formats for the two memory configurations considered (1,048,576 and 524,288.

The length of time required for data storage playback is determined by the downlink data rate. At the end of the orbiter mission, this data rate is 128 bps. At times other than at periapsis, the science data requirement

TABLE 4-16. CORE DATA STORAGE CAPACITIES

Total Words	Word/Memory Size			
	8 Bit	16 Bit	24 Bit	32 Bit
2,048*	-	-	49,152	65,536
4,096*	-	65,536	98,304	131,072
8,192*	65,536	131,072	196,608	262,144
16,384*	131,072	262,144	393,216*	524,288
24,576	196,608	393,216	589,824	786,432
32,768*	262,144	524,288	786,432	1,048,576

*Binary Multiples — best and most effective.

TABLE 4-17. DATA STORAGE IMPLEMENTATION REQUIREMENTS

Requirements	Memory Configuration	
	1,048,576	786,432
Bit rates	8-2048 in 2^n min plus 640 bps	Additional rates of 170-2/3, 341-1/3, and 10-2/3
Formats	No additions	Possible addition of a single periapsis format

is approximately 8 bps. Allowing another 20 bps for engineering data transmission, etc., the remaining capability of 100 bps is available for dumping data from the storage unit. The length of time required for playback at this rate is approximately 3 h. Playback in this time interval is attractive as it allows complete playback of all stored periapsis data subsequent to periapsis and within the same Goldstone pass. This feature allows consideration of deletion of some non-Goldstone tracking periods (Canberra or Madrid) as all mission critical activities can be conducted within a single Goldstone pass.

Utilization of this data storage unit during the time that Goldstone is out of view will assure that important scientific data is not lost in the event that reduced DSN tracking is required. The period of time between Goldstone rise and the start of periapsis activities is sufficient to play back all data recorded during the apoapsis period.

The requirements analysis arrived at the following conclusions:

1) The recommended data storage size is 1,048,576 bits. This size is attractive as it meets data storage requirements, with adequate overhead, for the periapsis and apoapsis phases. It is also a size that is desirable from a hardware implementation standpoint in that there are half capacity units (524,288 bits) available as existing hardware; spacecraft hardware is minimized as additional bit rates and formats are not required.

2) For a small increase in spacecraft hardware, an alternate memory configuration of 786,432 total bits is acceptable.

3) Use of the data storage at apoapsis would allow reduction in DSN tracking requirements and consequent cost savings.

Hardware Implementation Analysis

This subsection reports a tradeoff analysis of data storage technologies for the orbiter spacecraft. The technologies reviewed were magnetic core, magnetic plated wire, semiconductor bipolar, and semiconductor static and dynamic MOS. Magnetic tape recorders were not considered due to the moderate storage requirement (≈393 K bits) and because of the high mass, low reliability and constant change in center of gravity normally associated with their use. The basis for the conclusions in this study are that cost, mass, and reliability are of prime importance. Other items of significant importance are power, magnetic cleanliness, and use of existing technology.

For data handling memories in the order of 10^5 to 2×10^6 bits, either the use of a magnetic core memory or a MOS dynamic memory with a hidden refresh cycle is feasible. However, magnetic core was chosen for use on Pioneer Venus because the technology has been proven on past space programs and thus offers the best approach in terms of technical risk. Table 4-18 summarizes some of the parameters of the different types of memories investigated in this study.

The various memory technologies investigated in this study are briefly discussed in the following sections.

Magnetic Core. Memories consisting of magnetic cores are usually constructed in two basic sections, electronics and core stack. The core stack is usually a three-dimensional array organized in stacked planes. The electronics usually account for about 50 percent of the volume and may be considered to fall within three major categories:

1) Address electronics (X and Y)

2) Bit electronics (Z) (read and write)

3) Power control (strobing and conversion)

A major feature of core memories is that they are nonvolatile and therefore can be power strobed only when access is needed. The typical access time is from 1 to 3 μsec. The power during access is very large and in the order of 50 W. If the memory is operated at low data rates (i.e., less than 100 K bps), it can be organized such that access is once every 20 to 40 bits and therefore the average power consumption could be less than 1/2 W.

Magnetic cleanliness requirements may present some moderate problems to magnetic core memories. If the requirements are not too severe (i.e., in the order of 5 gamma at 3 ft), careful packaging techniques and selection of materials can be used to reduce the magnetic fields to acceptable levels.

TABLE 4-18. SUMMARY OF DATA STORAGE TECHNOLOGIES

Parameter	Magnetic		Semiconductor		
	Core	Plated Wire	Bipolar	MOS Static	MOS Dynamic
Nonvolatile	Yes	Yes	No	No	No
Power	Low	Lowest	Highest	High	Moderate
Mass	Moderate	Moderate	High	Moderate	Lowest
Space experience	High	Low	Low	Low	Low
Ability to achieve magnetic cleanliness	Moderate	High	High	Highest	Highest
Radiation tolerance	High	Highest	High	Moderate	Moderate
Technical risk	Low	Moderate	Low	Low	Moderate
Cost	Lowest	Highest	High	Moderate	Moderate
Potential availability of existing hardware	Highest	Moderate	Low	Low	Low

Size and mass of a custom designed random access core memory in the range of 100 K bits to 2 megabits, with 28 V power supply and using standard small cores, should fall in the range of 2.3 to 4.5 kg (5 to 10 lb) and require a volume between 820 to 2460 cm^3 (50 to 150 in^3). These quantities depend primarily upon the memory configuration and associated electronics, and to a much smaller extent on the quantity of storage bits.

Magnetic Plated Wire. Plated wire memories are constructed similar to core memories (i.e., the electronics are separated from the stack). Organization of plated wire memories tends to prefer long internal word lengths, compared to core, in order to have an efficient configuration. Some major features of plated wire memories are that they are nondestructive readout, nonvolatile, and require lower drive currents than core memories.

Magnetic cleanliness should be easier to achieve with plated wire than with core memories primarily because of the lower drive currents; however, design difficulties may be encountered if ultra magnetic cleanliness is required.

Size and mass should be about 80 percent of an equivalent core memory because both the stack and the lower current drive electronics can theoretically be smaller and lighter in weight. However, a plated wire memory may be larger and weigh more than an equivalent core memory, depending upon the particular memory configuration and organization.

The cost of a plated wire memory is currently two to three times that of an equivalent core memory. This is primarily due to a limited demand at the present time for this type of memory and the consequent lower production and manufacturing experience.

Bipolar Semiconductor Memories. Bipolar semiconductor memories can be constructed in almost any desired word length because the semiconductor chips usually consist of 64 to 128 words 1 to 4 bits in length. Semiconductor memories are volatile and must have power applied continuously in order to retain their data contents. Bipolar memory devices are usually relatively small (e.g., 256 bits). To construct a megabit memory using these devices, approximately 4000 chips would be required. Using an optimistic product mass of 3.2 kg per 1000 integrated circuits, it appears that a bipolar memory would be very heavy and in the order of 11 to 14 kg (25 to 30 lb). Because of the high mass, bipolar memories are not recommended at this time.

MOS Semiconductor Memories. MOS semiconductor memories can be divided into two general groups, static and dynamic. Static memories act like a standard storage device when power is applied, but they use more power and have a lower bit density than dynamic memory devices. Dynamic memories must be periodically refreshed in order to maintain their data contents. Because the packing density of a dynamic memory is at least twice as great as that of a static memory device and since the required refresh circuitry is minimal, a dynamic memory is preferred. Assuming a 2048 bit dynamic MOS chip, the mass of a megabit memory is calculated approximately as follows:

	kg	(lb)
Power supply	0.4	(1.0)
Memory, at 3.2 kg/1000 IC	1.6	(3.5)
Refresh, at 10 percent of memory	0.2	(0.4)
Totals	2.2	(4.9)

Power requirements of dynamic MOS memories running at slow access times is dependent on the refresh period. However, for a megabit memory operating at low data rates, the power is estimated to be approximately 1 to 2 W for the memory and approximately 0.5 W for the refresh circuits. When the inefficiency of the power supply is included, the total power for this type of memory is estimated to be about 4 to 5 W.

Magnetic cleanliness requirements are easily met for semiconductor memories because of the low operating currents.

Environmental constraints for MOS dynamic memories are usually limited by radiation considerations. Radiation tolerance for these parts is typically 10^4 to 10^5 rads (S_i) of radiation before failure.

Probe Data Storage Hardware

This section analyzes the types of read/write memory for data storage use on the probes. Because of the considerable differences in storage requirements between the orbiter and the probes (approximately 393 kilobits for the orbiter compared to 4 kilobits for the large probe), a separate study was warranted; the study was initially reported in Reference 4-4.

The study determined that semiconductor read/write memories should be used on the probes, and that suitable memories already exist and are now being qualified for spacecraft use. However, the state of the art in semiconductor memories is advancing so rapidly that better devices may well be available by the time the final probe design is undertaken.

The Pioneer Venus baseline design requires that 4096 bits of data be stored in the large probe, during the entry deceleration, for transmission later in the descent; the small probes require storage of 512 bits of data. An investigation was conducted to determine the type of memory best suited to store these data. Factors considered in selection of a memory included cost, mass, power consumption, dimensions, availability, ability to survive the probe entry deceleration, radiation hardness, and the possibility of commonality between the large and small probe designs. The memory technologies considered included core, plated wire, bipolar semiconductor, and MOS semiconductor.

In general, magnetic memories (core and plated wire) are not optimum for small capacity applications such as the 512 and 4096 bit memories required by the probes. A magnetic memory in this size range (590 bits) was recently investigated for another Hughes space program, and the results of the investigation support the above conclusion. Table 4-19 contains the data from manufacturers' quotes on the 590 bit space qualified magnetic memory received during that investigation. The manufacturers indicated that the parameters would only vary a small amount with increases or decreases of a few times in the capacity, so these figures are felt to be

TABLE 4-19. 590 BIT MAGNETIC MEMORY CHARACTERISTICS

Memory Type	Dimensions, cm (in)	Mass (weight), kg (lb)	Power, W	Cost
Plated wire (Manufacturer A)	14.0 x 19.3 x 19.0 (5.5 x 7.6 x 0.75)	0.45 (1.0)	0.75	$250,000 nonrecurring $20,000 per unit
Plated wire (Manufacturer B)	14.7 x 20.3 x 3.3 (5.8 x 8 x 1.3)	0.90 (2.0)	0.35	$100,000 nonrecurring $5,000 per unit (without sufficient quality for Pioneer Venus)
Core (Manufacturer C)	10.3 x 15.2 x 1.3 (4 x 6 x 0.5)	0.22 (0.5)	0.5	$100,000 nonrecurring $5,000 per unit

a relatively accurate representation of magnetic memory parameters for the probe application. Because of their excessive size, mass, and cost, these memory types were not selected for the probe data memory.

Data on a variety of semiconductor random access memories (RAMs) were also collected. The objective of the investigation was to identify a memory device capable of meeting the needs of the probe data memories, and to generate data for determining probe characteristics based upon the use of this device. However, the state of the art in semiconductor memories is advancing so rapidly that in all probability, by the time of the Pioneer Venus final design, better devices will be available for spaceborne use than there are at the present time. The characteristics of several currently available RAMs are given in Table 4-20.

Upon examining the characteristics of the available semiconductor memories, it was observed that memories containing more bits do not necessarily dissipate more power than memories with fewer bits. This is due to the fact that the maximum heat that can be dissipated in an integrated circuit package tends to be a constant limiting factor upon designs. Thus, the designer does not necessarily pay a penalty of higher power for choosing a memory device having more bits. Therefore, in order to utilize common devices, it would be preferable to use memory chips of 1024 or 2048 bits

TABLE 4-20. CHARACTERISTICS OF SOME SEMICONDUCTOR RANDOM ACCESS MEMORIES*

ORGANIZATION WORDS X $\frac{\text{BITS}}{\text{WORD}}$ = BITS	MANUFACTURER	PART NO.	TECH.	STATIC OR DYNAMIC	POWER (mw)	SUPPLY VOLTAGES	DT/TTL COMPATIBLE	SPACE QUALIFIED
16 x 1 = 16	Texas Instruments	SN5481	Bipolar	Static	300	+5V	Yes	No
16 x 4 = 64	Advanced Micro D. Computer Microtech.	AM31L01 CM2106	Bipolar	Static	135	+5V	Yes	Yes
64 x 2 = 128	Collins Radio	CRC 4002	MOS-P	Static	150	+12V	Yes	No
128 x 1 = 128	Advanced Mem Sys	AMS1503	Bipolar	Static	400	+5V	Yes	No
128 x 2 = 256	Motorola	MCM 4257	Bipolar	Static	500	+5V	Yes	No
64 x 4 = 256	Collins Radio	CRC 4003	MOS-P	Static	300	+12V	Yes	No
256 x 1 = 256	Intel	1101A1	MOS-P	Static	200,400	+5V, -9V	Yes	No
256 x 1 = 256	Computer Microtechnology	CM2150	Bipolar	Static	300	+5V	Yes	Yes
256 x 1 = 256	Solid State Scientific	SCL5553	C-MOS	Static	.060	+12V	Yes	No
1,024 x 1 = 1,024	Electronic Arrays	EA1502	MOS-N	Dynamic	165	-12V, +12V	Yes	No
1,024 x 1 = 1,024	Intersil	IM5508	Bipolar	Statis	100	+5V	Yes	No
1,024 x 1 = 1,024	Mostek	MK4006P	PMOS	Dynamic	450,50	+5V, -12V		No
1,024 x 1 = 1,024	Fairchild	93415	Bipolar	Static	500	+5V	Yes	No
1,024 x 1 = 1,024	Raytheon	R5500	Bipolar	Static	400	+5V	Yes	No
1,024 x 1 = 1,024	TI	SN54S204	"	Static	500	+5V	Yes	No
1,024 x 2 = 2,048	TI	TMS4020	P-MOS	Dynamic	320	+2V, -16V	Yes	No
2,048 x 1 = 2,048	Advanced Memory Systems	AMS6003	MOS	Dynamic	164	+5, +8, -15V	Except Clock	No

*The memories described do not represent a complete listing of available devices. Emphasis was placed upon devices in the size range appropriate for the probe data memory. Where two values of power are given for a device, the lower value represents a "standby power" sufficient to maintain the data in memory, while the higher value is an "operating power" necessary to load or interrogate the memory. Memories are listed as "space qualified" if qualification procedures are now complete or under way and are expected to be complete before the conclusion of the present Pioneer Venus study contract. Even "space qualified" memories may require additional testing to verify survival of the probe 500 g deceleration, however.

in the design of both of the probe data memories if a qualified device were available. Even though the small probe requires only 512 bits, using this approach to commonality is more attractive than using 256 or 512 bit memory chips in the large probe.

However, since space qualified 1024 or 2048 bit devices are not presently available, the memory chosen for use on both probes is the Computer Microtechnology CM2150, a 256 bit bipolar device because it is being qualified by Hughes for another space program. Thus, the qualification cost associated with use of this memory device on Pioneer Venus will be limited to the cost of verifying survival of the Venus entry deceleration.

On the other hand, since 16 CM2150 devices are required to build the large probe memory, and since the power dissipation of the device is also relatively high, it still might be preferable, in spite of qualification cost saving, to consider use of an alternate device. Several lower power and higher capacity devices such as the Solid State Scientific SCL 5553, a 256 bit CMOS memory dissipating only 60 microwatts or the Intersil IM 5508, a 1024 bit bipolar device with one-twelfth the power dissipation per bit of the CM2150, will continue to be monitored in the event that one should become qualified.

Probe Stored Data Playback Techniques

This subsection reports the results of a study upon the probe data handling subsystem of employing a single, uninterrupted memory "dump" to return the data stored during entry deceleration as opposed to interleaving it with real time science data transmission; the initial report of this study may be found in Reference 4-4.

It was found that, contrary to early expectations, playing back the probe stored data in a single "burst," which would interrupt the transmission of real time data, did not allow a power reduction in the probes. Because no advantages of a burst playback were found, and because of the additional complexity of this technique, the approach of playing back the stored data interleaved with real time data is preferred.

Hughes original design employed interleaved playback, as recommended by the science definition study. However, the possibility of a power saving was thought to exist if a dump approach were used, in that the data memory can be powered down after the completion of stored data transmission; a dump might allow this to be accomplished sooner. The investigation was conducted using current estimates of probe descent profiles and the science data rate requirements given in the preliminary science definition report. Both interleaved and burst playback were found to be feasible.

The interleaved data playback was chosen; following are the considerations which led to this decision:

1) The use of "dump" playback would require an additional data format, which would complicate the data handling subsystem as well as ground processing.

2) The use of a data dump would not, in fact, allow the probe data memory to be powered down appreciably sooner than would be the case with interleaved playback in a properly planned format. If a memory dump were conducted during the early phases of the descent, capability would be required to either store data during the dump, or to interrupt the memory dump frequently for long periods of time to accommodate real time data transmission. The store during dump capability would complicate the probe data handling subsystem appreciably and would require that the memory remain powered until the data stored during the dump were played back. The data interrupts required by this approach would be so frequent and so lengthy as to essentially amount to interleaved playback. If the memory dump were delayed until just before parachute release, it might be possible to complete an entire memory dump playback between required sample points of real time data. However, present figures for data rate and descent rate indicate that this would not be possible without increasing the transmission bit rate. Furthermore, delaying the memory playback until this time would require that the data memory remain powered longer than would be the case if interleaved playback were used.

3) Interleaving of stored and real time data has the disadvantage that words in the format, used for data playback, would contain redundant information after the memory had been played back once. Since the percentage of bandwidth used for playback is small, this is not a serious disadvantage. The present figures indicate that a 147 word format, including six words of stored data playback, would accommodate all large probe experiments. This format, at 256 bps would allow two complete playbacks of the stored data in less than seven minutes. After that time, the memory could be powered down. If the playback were conducted as slowly as possible, while still achieving two complete playbacks of data before the probe reached an altitude of 20 km, less than 3 percent of the data format would be devoted to stored data playback in the large probe. In the small probe, because of the more rapid descent, the stored data playback would occupy approximately 5 percent of the data bandwidth. In both the large and the small probes, it is possible to play back the stored data interleaved with real time data without requiring a major percentage of the data bandwidth. Two data playbacks could still be completed early enough in the descent to allow important power savings by powering down the data memory.

Probe Multiple Data Formats

This subsection reports on an investigation of the impact upon the probe command and data handling subsystems of providing two or more data formats for use during different phases of the probe entry and descent; the initial report of the study may be found in Reference 4-4.

The additional cost to the probes of providing more than one data format is not expected to be appreciable. This results from the fact that the probe data formatting is controlled with read only memory (ROM) elements. Depending upon the complexity of the additional data formats, which will not be fully defined until system design is completed, it is expected that the cost will be limited to providing the additional command outputs necessary to change formats plus some additional ground processing complexity. It is felt that the overall system advantage to be gained by providing multiple formats (i.e., to meet science data return requirements without having to add increased rf power amplification) is worth the possible increase in complexity.

An investigation was made to determine the modifications required in the probe data handling subsystem to accommodate more than one data transmission format. In the initial design for the probes, ROM devices were used in order to accommodate a relatively complex data format and to allow changes to the format until as late in the program as possible. Since the optimum way to design for multiple data formats is also by use of ROM devices, the existing implementation allows them to be accommodated with a minimum of impact on the design.

During the initial design, a survey of ROM devices was conducted. The Harris HPROM 1024, a 1024 bit bipolar TTL compatible device, was selected. A combination of these devices was used to implement the design; this may be adequate to control more than one format, depending upon the total number of data inputs, the sampling requirements of each, and the method of implementing the programming of data formats. The most cost effective implementation cannot be determined until final definition of the data formats is complete. However, it is felt that the flexibility offered by the use of ROMs will allow provision for multiple data formats without significant impact upon the data handling subsystem design. Command subsystem outputs are, of course, needed to change the formats. But sufficient spare outputs are presently available to accommodate this without change to the command subsystem design. Additional formats will, to some extent, increase the complexity of ground processing equipment and/or software.

The savings afforded to the overall system design by the use of multiple formats appear to justify the slight increase in complexity to the data handling subsystem. The Harris memory is preferred, in spite of its relatively high power dissipation, because it is presently being qualified for space programs. It is possible that an even better device may be qualified by the time of the final probe design; but a change of memory should not affect

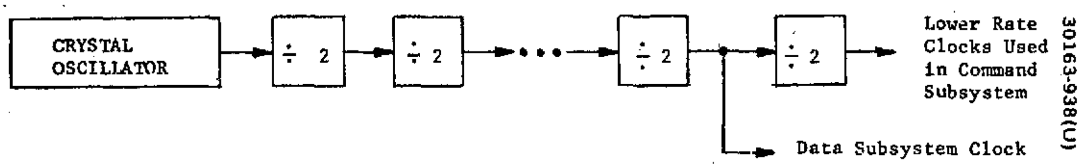


FIGURE 4-9. SINGLE DATA RATE SYSTEM

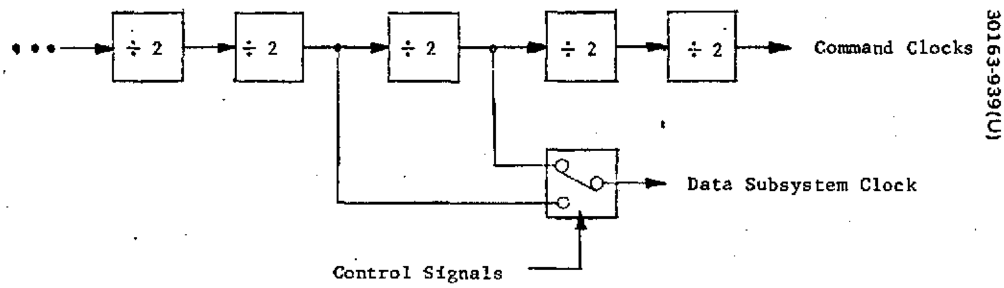


FIGURE 4-10. MULTIPLE DATA RATES IN BINARY RATIOS

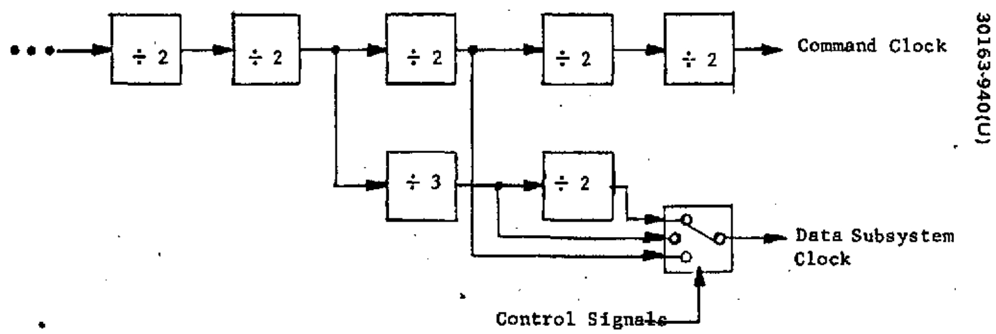


FIGURE 4-11. MULTIPLE DATA RATES IN NONBINARY RATIOS

the result of this study. The only advantages of using another memory would be a reduction in memory power dissipation. In the present probe design, the power dissipation of the ROM has been reduced to a very low average value by switching power to the ROM only when it is to be interrogated. Therefore, the Harris ROM can be used with no appreciable power or weight penalty, and additional probe data formats can be generated without appreciable changes to the initial design.

Probe Multiple Data Rates

Because of the attenuation of the Venusian atmosphere, power in the communications subsystem can be minimized by transmitting data more slowly at lower altitudes, and a rate reduction appears feasible from a systems standpoint. The subsection reports a study to determine the weight and power impact on the probe command and data handling subsystems from the inclusion of more than a single data rate. As expected, the weight and power increases were small. The use of multiple data transmission rates, including certain rates which are not related by integral binary ratios, does not significantly complicate the probe electronics. Details of this study were initially reported in Reference 4-5.

To determine the design impact of providing multiple data rates, circuit designs of the relevant portions of the command and data handling subsystems were prepared. The designs incorporated provisions for various data rates and combinations of rates. Rates that are not related by integral power of two ratios were included.

In the initial design, the data handling subsystem of the probe is operated from a single clock frequency. This single clock drives the timers, multiplexers, data formatter, convolutional coder, and all other elements of the data handling subsystem. It is gated and provided to the science instruments as a read clock. Thus, the probe data rate can be varied, without altering the data format, by simply varying the subsystem input clock rate. Furthermore, the mass, cost, and volume penalties in the command and data handling subsystems due to providing more than one data rate are entirely associated with the added circuitry needed to generate and select the required clock frequencies.

A common crystal oscillator and countdown chain provides clock and timing pulses for the entire command and data handling subsystem of the probe. If only a single data rate were required, this countdown chain would be implemented as a series of divide-by-two stages as shown in Figure 4-9. With the addition of gating for clock selection, as shown in Figure 4-10, this same arrangement of counters can provide multiple data rates as long as the required rates are related by ratios that are integral powers of two. Rates between the integral powers of two can conveniently be obtained by use of a divide-by-three stage (or a divide-by-five, -seven,...) followed by any additional stages necessary to provide data subsystem clock rates; this method is illustrated in Figure 4-11.

Considering the single data rate case (Figure 4-9) as a basis for comparison, the increments to parts count, mass, and power were computed for: 1) systems providing two data rates related by an integral power of two ratio, 2) for a system providing two rates not related by an integral power of two, and 3) for a system providing up to four rates not related by powers of two. The estimates are based upon the use of low power transistor/transistor logic (TTL) circuits, and represent realistic increments to the subsystem design.

Table 4-21 presents the results of the study. The quantity of circuitry added to provide a multiple data rate capability in the probes is very small, even if rates differing by ratios other than integral powers of two are chosen. The mass, parts, and power increases in no case exceeded 4 percent. Therefore, the conclusion is that multiple data rates should be used in the probes if their use will significantly reduce the mass or power of the other probe subsystems. This conclusion supports the initial decision to provide for a reduction of the data rate at low altitudes to one-half its value at higher elevations.

When more than one data rate is to be used, command outputs must be utilized to select the desired rate. However, sufficient spare outputs are available to accommodate this data rate selection.

TABLE 4-21. IMPACT OF MULTIPLE DATA RATES

Multiple Data Rate Alternatives	Mass (weight) Increase kg	Mass (weight) Increase (lb)	Power, Increase, mW	Component Increase ICs
Choice of two rates related by binary ratio	0.014	(0.03)	10	2
Choice of two rates related by $1{:}2^n/3$ ratio	0.023	(0.05)	20	3
Choice of up to four rates related by binary and by $1{:}2^n/3$ ratios	0.045	(0.10)	45	6

4.3 SURVEY OF AVAILABLE SUBSYSTEM HARDWARE

Minimum cost and risk should result from using existing, proven designs and hardware for the command and data handling subsystems. Accordingly, known suppliers of space command and data handling equipment have been solicited to determine the availability of equipment which could meet program requirements with no more than minor modifications to existing designs.

Preliminary specifications of requirements on the command and data handling subsystems for the Pioneer Venus spacecraft and probes were prepared as a basis for solicitation of data from vendors. As much as possible, parameters were specified loosely or not defined in order to enable existing hardware to meet the requirements. During December 1972, 12 companies with related experience were invited to submit technical and rough order of magnitude cost data. Six companies responded with data: one on command equipment only, one on data handling equipment only, and four on both. In addition, some of these and other companies provided verbal information in response to our telephone or in person contacts.

In addition to outside sources, command and data handling equipment designed by Hughes was reviewed for applicability to Pioneer Venus.

Spacecraft Command Subsystem

Analysis of the available command equipments has led to a buy decision for the command demodulator and a make decision for the decoding equipment using OSO-I designs.

Communications link considerations have dictated the desirability of using a phase-shift-keyed (PSK) demodulator which provides a theoretical 3 dB improvement over the more commonly used frequency-shift-keyed (FSK) demodulator. Spaceborne PSK demodulators with characteristics generally suitable for Pioneer Venus are under development by several companies. For example, two such demodulators are under development by RCA and Motorola for the Viking Lander and the Viking Orbiter, respectively. Hughes, on the other hand, does not have a PSK demodulator with characteristics as suitable for Pioneer Venus as the above designs. The demodulator, therefore, is recommended to be a buy item.

None of the decoders for which vendor data was obtained appeared to be very attractive for the spacecraft decoding application. All vendors have to modify existing designs or perform a new design. Analysis of the limited data received indicates that, of the vendors who responded, Radiation, SCI Systems, and Texas Instruments have designs which could be modified to provide an acceptable combination of characteristics.

Review of several Hughes decoder designs indicates that the command decoding system for the OSO-I program is not only the best choice among

Hughes designs, but also has a number of distinctly advantageous features. Among these are:

1) The command output module, that interfaces with science instruments and spacecraft subsystems, has been carefully developed over a decade and represents an optimum interface from the points of view of noise tolerance and inadvertent command execution due to faulty equipment. In addition, its use further enables use of some other subsystems from OSO-I, Hughes' most current science exploration satellite, with little or no modification. Also, it is an interface with which Hughes' science integration and spacecraft development engineers are intimately familiar.

2) The OSO-I command subsystem is modular, thus providing flexibility to accommodate larger or smaller command requirements than in OSO-I by simply adding or deleting command output modules. This feature has proven to be very desirable on several programs using a similar modular concept. It permits completion of command subsystem hardware design and fabrication before completion of a final specification on total command requirements, which frequently vary during a spacecraft development phase.

3) The distributed nature of the OSO-I subsystem, in which the command output modules (remote decoders) can be located close to user subsystems and science instruments, results in a much simpler harness management job than when a centralized decoding subsystem is used. This is because only one signal and one power line (plus ground) connect each of two redundant central decoders with each command output module, each of which have an output capability of 64 pulse commands and four serial magnitude commands. This may also provide a weight reduction, since the weight of the additional electronic parts used in the modular subsystem are more than compensated by the reduction in wire weight of the command output lines when the average distance between central decoders and user subsystems exceeds 1 m.

4) The OSO-I command output module (remote decoder) can be used without modification for Pioneer Venus.

5) The OSO-I command output module uses custom MOS-LSI devices to mechanize most of its functions, which permits inclusion of several desirable features while minimizing weight. The Hughes MOS-LSI devices are built using a space process that can tolerate a radiation dose of 10^5 rads (Si), which is greater than the anticipated requirement during the longest orbiter mission.

For these reasons, the existing OSO-I command output module is recommended for the Pioneer Venus spacecraft.

A new central decoder design is recommended; the unit will be designed to provide compatible interfaces with the PSK demodulator to be selected as well as with the existing command output module design. Much of the circuitry would be the same or similar to that in the OSO-I central decoder. In addition, the receiver reverse switching function and the command memory function would be included in this unit.

Spacecraft Data Handling Subsystem

Analysis of the available data handling equipments has led to the following decisions: 1) a make decision for the data input module and PCM encoder unit, both of which use unmodified OSO-I designs; 2) a buy decision for an available, or slightly modified, magnetic core data storage unit (orbiter spacecraft only); and 3) a make decision for a new telemetry processor which provides a compatible interface with each of these existing designs.

As was the case with command decoders, no vendor of data handling equipment who supplied information in response to Hughes request has an existing design, which is close to meeting the principal requirements for the Pioneer Venus spacecraft. While data received were somewhat inadequate for a thorough assessment, it appears that each of the equipments was deficient in one or more respects. Some of these deficiencies are as follows:

1) Only one or two data modes (formats) are provided. As many as seven may be necessary for both the multiprobe spacecraft and the orbiter spacecraft, with simultaneous real time and stored formats required for the latter.

2) Too few data inputs were provided.

3) No serial digital data input capability (e. g. , radiation).

4) Less than 8 bit encoding (e. g. , Texas Instruments and TRW).

5) In one case (SCI systems) a new design was proposed with no reference to a specific related existing design.

Of Hughes telemetry equipment designs, the OSO-I design is most applicable to Pioneer Venus. The OSO-I telemetry and data handling subsystem has two units that meet all Pioneer Venus requirements. These are the data input module (remote multiplexer) and the PCM encoder unit. Some of the desirable features of these units are:

- The data input module provides a data interface with science instruments and spacecraft subsystems that accommodates analog, serial digital, and discrete bilevel data in several pin-programmable mixes depending on specific user-subsystem requirements.

- Any data input line can be shorted to a voltage between -50 and +50 V with no damage or degradation of other channels. This protects hardware from possible damage during system test, eliminating a potential schedule risk.

- The OSO-I telemetry subsystem is modular, thus providing the same flexibility to accommodate subsystem changes as cited above for the OSO-I command subsystem.

- The distributed nature of the OSO-I subsystem, with data input modules located near user subsystems, also results in a much simpler harness management job and a potential weight savings; the advantages are similar to those cited for the command subsystem above.

- Each data input can be addressed in a random access manner. Data formats are, therefore, completely under the control of a format generator in the telemetry processor. Since data transfer between the data input module and the PCM encoder and format generator is by means of high speed data bursts, simultaneous stored, and real time formats can be easily implemented.

- Both units can accommodate an interrogate channel rate of dc to 3200 channels/second; this can easily meet the needs of Pioneer Venus. The channel and bit rates are completely controlled by the format generator.

- Like the command output module, the data input module uses modern MOS-LSI technology to achieve its functional capability with minimum weight and maximum reliability.

- The PCM encoder converts analog data to the required 8 bit resolution. Accuracy is enhanced by differentially encoding each data point relative to the analog signal ground in the user subsystem. The encoder also feeds a precision current via the data input module to a user's resistive transducer (such as a thermistor temperature sensor), while the latter is addressed. This current is used as a stimulus to produce a voltage which can be measured.

The OSO-I data input module and PCM encoder are therefore recommended for the spacecraft data handling subsystem, since they appear to be highly suitable for the Pioneer Venus spacecraft application. Furthermore, they provide interfaces with science instruments and spacecraft subsystems with which Hughes' science integration and spacecraft development engineers are intimately familiar and which permit the use of a number of existing interface circuit designs in the user subsystems.

The orbiter data storage hardware trade study (see subsection 4.2) recommended the use of a magnetic core memory for the data storage application. Several vendors have developed, or are developing, spaceborne core memories having capacities in the range required for Pioneer Venus; good potential exists for one of these memories being usable as designed or with minor modifications. Among the candidate vendors for core memories are Electronic Memories (Viking Lander, SEMS-8 or SEMS-9 designs), SCI Electronics, Spacetac, Standard Electrik Lorenz (Helios design), and Fabri-Tek. If cost is comparable, a plated wire memory would be acceptable; possible vendors of such memories are SCI Electronics, Motorola, Honeywell, and Sperry Rand. As of this writing, these vendors have been requested to provide a technical proposal and cost estimates for a memory meeting the orbiter data storage requirements.

A telemetry processor is required to perform the format generation, timing, convolutional encoding, and modulation functions; its interfaces must be compatible with the selected data storage unit, the PCM encoder, the data input module and the rf subsystem. This unit would have to be of new design. Many of the circuits for the format generator and timer, however, are available from the OSO-I format generator and spacecraft clock units. Design by Hughes is recommended because of the applicability of this OSO-I circuitry and because of the need for close interface coordination with designers of the other units, particularly the PCM encoder and data input module.

Probe Command and Data Handling Subsystem

No existing command and data handling hardware has been found that has been designed to meet the 500 g peak deceleration. This would dictate a new or modified packaging design independent of whether circuit designs are of existing or new design. The probe equipment, unlike the spacecraft equipment, is also subject to severe volumetric and form factor constraints. Hughes feels that this constraint makes mandatory a new packaging design by product design engineers working in close coordination with probe integration personnel. While existing or modifications of existing circuits may be used, this constraint also dictates a circuit design tailored to the probe requirements. Since an off-the-shelf design that is adequate is likely to have features and circuitry greater than required, its volume and weight are also likely to be excessive.

For these reasons, Hughes recommends an in-house design of the probe command and data handling subsystems. While a new packaging design using established techniques is required, a number of circuits from OSO-I are applicable with some modification. Electronic parts are mostly of the same types already space qualified and planned for use on the Pioneer Venus spacecraft.

Pyrotechnic Control

While the functional complexity of the pyrotechnic control circuitry (squib drivers) is much simpler than the remainder of the spacecraft and probe command and data handling subsystems, the irreversible nature of squib firings necessitates a design which is both highly reliable and very insensitive to input signal and power line noise. Such circuitry has been successfully used on all Hughes spacecraft. In particular, the OSO-I squib driving circuitry has been designed to be compatible with the command interface recommended for Pioneer Venus. However, the requirement to fire three squibs simultaneously in the Pioneer Venus space vehicles necessitates use of a higher power output transistor. The same circuit will be used for spacecraft and probe applications.

4.4 PYROTECHNIC TRADE STUDY

Actuation of mechanical components such as separation nuts is required on the orbiter, probe bus, and probes. Pyrotechnic mechanics will be used to activate or trigger these components.

A tradeoff has been made between alternate designs in order to determine pyrotechnic configurations. The three basic elements of a pyrotechnic subsystem analyzed were:

1) Initiator characteristics, i.e., explosive and nonexplosive types.
2) A means for providing a highly reliable method of power switching to provide electrical power to the specific initiation selected.
3) A power source capable of providing a large current to an initiator for a short time period.

Initiator Characteristics

Two basic types of initiators are used. A bridge wire type of initiator (squib) ignites a small amount of propellant and generates gas at high pressure. This is generally used to actuate separation nuts, pin pullers, experiment breakoff hats, valves, orbit insertion motor igniters, and other functions.

The other types of initiator is a nonexplosive hot wire initiator which ruptures a hot wire element that pulls a pin. This type is used to disengage electrical connectors, jettison windows on the large probe, and deploy mechanisms on the small probes.

Bridgewire Types

Actuation time following application of current to squibs is not critical to most users. A 50 to 100 ms firing pulse is normally applied to guarantee firing, although most squibs in less than 5 ms. However, separation of the small probes from the probe bus by means of three separation nuts must be timed very accurately. This requires an accurate determination of the delay time to properly time the firing pulse. It also requires a high degree of simultaneity between the three separation nuts. Separation of the large probe from the probe bus and parachute and aeroshell release from the large pressure vessel also require good simultaneity. Because release from the large pressure vessel also require good simultaneity. Because of these timing requirements, the power switches (squib drivers) that deliver the firing current to the squibs must also operate simultaneously.

During firing the bridgewire initiator cartridge contains ionized gases that can provide a low resistance path from pin to pin and pin to case. Precise quantitative data are not available; therefore, it must be assumed that during the nominal 50 ms firing pulse each squib can present a hard short circuit to the squib driver. This presents potential problems that will be discussed below.

Most squibs have a very large postfire pin to pin and pin to case resistance after combustion has been completed. However, occasionally expended squibs have a relatively low postfiring resistance of approximately 50 ohms. This makes it mandatory that the squib be disconnected from the power source after the firing sequence is completed in order to remove a potential parisitic load.

The Single Bridgewire Apollo Standard Initiator (SBASI) has been selected for Pioneer Venus. The key performance parameters are:

• All fire	3.5 A with 0.999 reliability over -260° to +300°F
• No fire	1 A/1 W for 5 min
• Bridgewire resistance	0.95 to 1.15 ohms
• Functioning time	Approximately 2 ms to reach peak pressure at recommended firing current of 5 A

Hot Wire Initiators

Hot wire initiators are miniature, nonexplosive pin pullers. They have a wire element wound on helical grooves of a nonconductive, axially split spool that bears against a spring loaded plunger. The wire element has a short, reduced diameter section that offers local, increased electrical resistance. When the required current (3.5 A minimum) is passed through

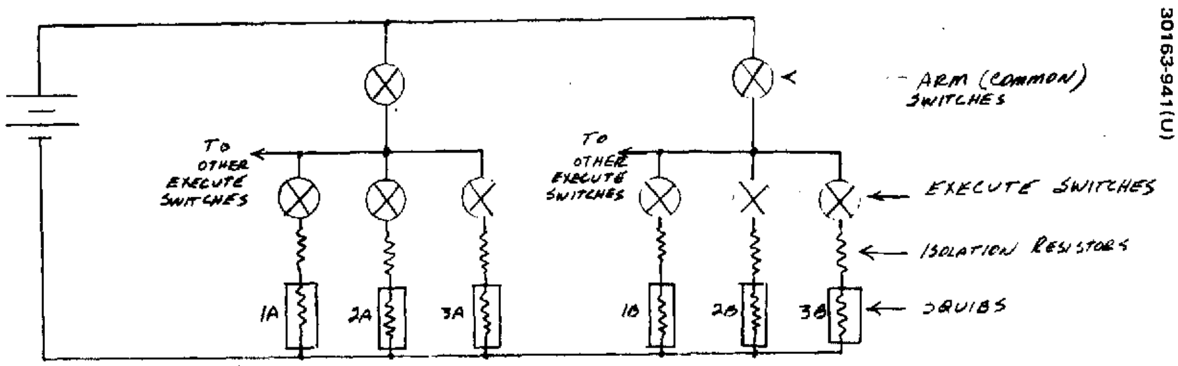


FIGURE 4-12. SQUIB DRIVER CONFIGURATION A

the wire, the reduced section heats rapidly to a temperature where the tensile strength is reduced to the point of rupture. The remaining wire unwinds and springs outward from the spool, permitting the spool halves to be separated by the wedging action of the spring loaded plunger. The plunger continues to be driven by the spring force, withdrawing the exposed end into the initiator housing.

Electrical characteristics of the wire element are similar to squib bridgewires. They have a nominal resistance of 1 ohm and will fire within 11 ms at 4.5 A. No fire rating is 0.5 A for 5 min.

Squib Drivers

General Squib Driver Requirements

The squib drivers must supply a minimum of 5 A to each bridgewire or hot wire for approximately 5 ms. Since pyrotechnic actions are irreversible, inadvertent actuation must be prevented by the following precautionary steps:

1) An arm switch plus an execute switch must both be closed before firing current is delivered to a squib.

2) The execute command must follow the arm command within 100 ms, otherwise the circuits are disarmed.

The latter features means that all firing commands must be provided to the squib driver from command memory. A number of different driver configurations were investigated as described below. The most demanding requirement is to fire six squibs (three primary and three backup) with a high degree of simultaneity. Therefore, this will serve as a common point of reference in a tradeoff of four different configurations.

Individual Squib Drivers (Configuration A)

Figure 4-12 shows a configuration derived from OSO. Squibs 1A and 1B are redundant; the same is true of 2A and 2B and also 3A and 3B. The upper (arming) switches are common, but each squib has its own individual lower (execution) switch. Each execute switch has a series isolation resistor to prevent excessive battery current in case of squib or harness short. It also prevents shunting of current from the other squibs by the shorting squib and thus, insures good simultaneity.

The probe bus, orbiter, and four probes have a total of 79 bridgewires and hot wires. This configuration would meet all their requirements satisfactorily, but it would require a very large number of power transistors. This, in turn, would result in a heavy and large pyrotechnic control unit (PCU).

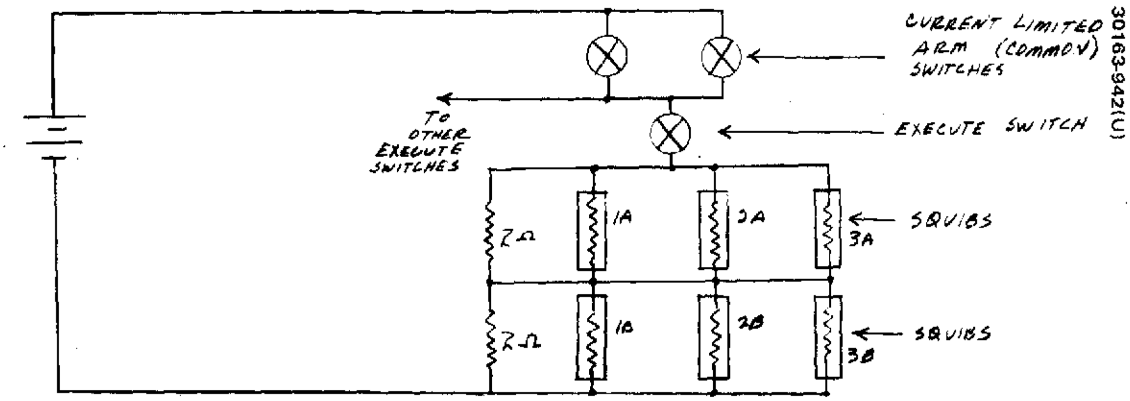


FIGURE 4-13. SQUIB DRIVER CONFIGURATION B

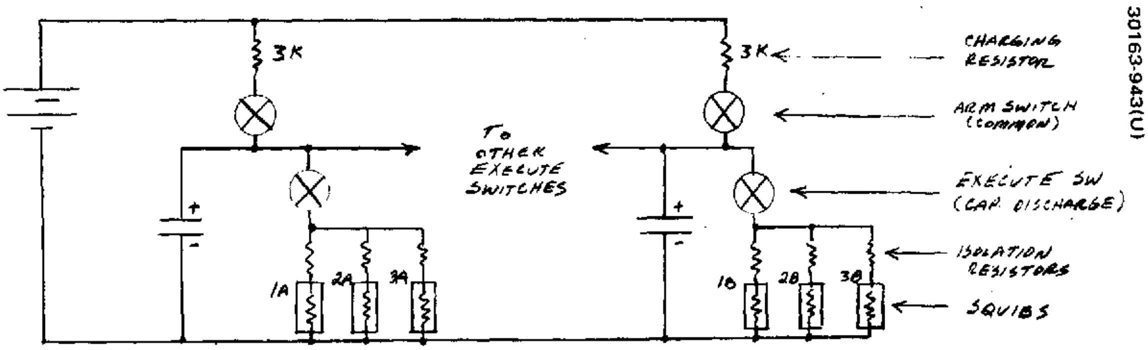


FIGURE 4-14. SQUIB DRIVER CONFIGURATION C

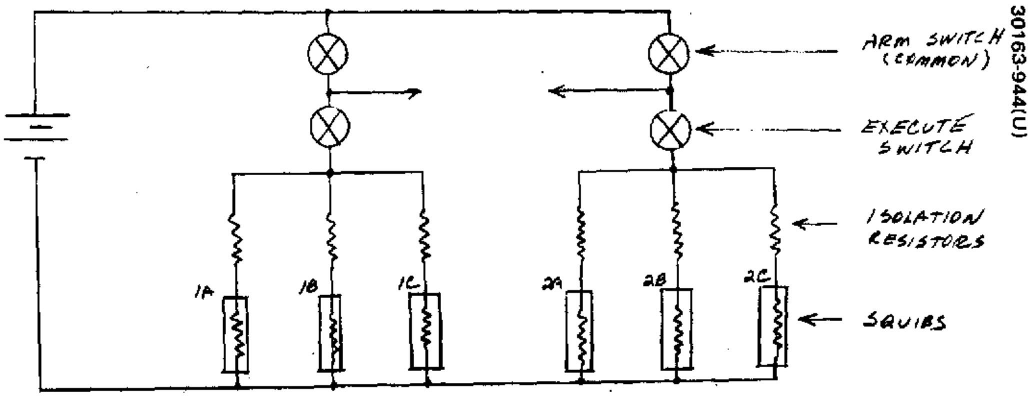


FIGURE 4-15. SQUIB DRIVER CONFIGURATION D (BASELINE)

Squib Driver Firing Series/Parallel Network (Configuration B)

Figure 4-13 shows a configuration that was successfully utilized on Surveyor to actuate three pyrotechnic devices simultaneously. The redundant upper switch provides current limiting in the event of squib shorts. This switch pair is common to all squib networks. Each squib network is actuated by one common execution switch. Squibs 1A and 1B are redundant and activation of either one will successfully perform the required function. Squibs 2A and 2B, 3A and 3B provide redundancy in the same way. The 2 ohm resistor provides a shunt bypass to compensate for open failures.

The most attractive feature of Figure 4-13 is that one power switch fires a large number of squibs. However, it does not have execution switch redundancy and a short during firing in any of the squibs could result in poor simultaneity.

Capacitor Discharge Squib Driver (Configuration C)

Figure 4-14 shows a configuration that has been used on Mariner spacecraft. It utilizes a large capacitor that slowly charges through a series resistor and arm switch. The squibs are fired by an execution pulse that discharges the capacitor into the squibs and isolation resistors. The advantages of this configuration are:

1) Low battery current during capacitor charging

2) Good simultaneity due to high initial peak currents

3) Low number of squib drive switches

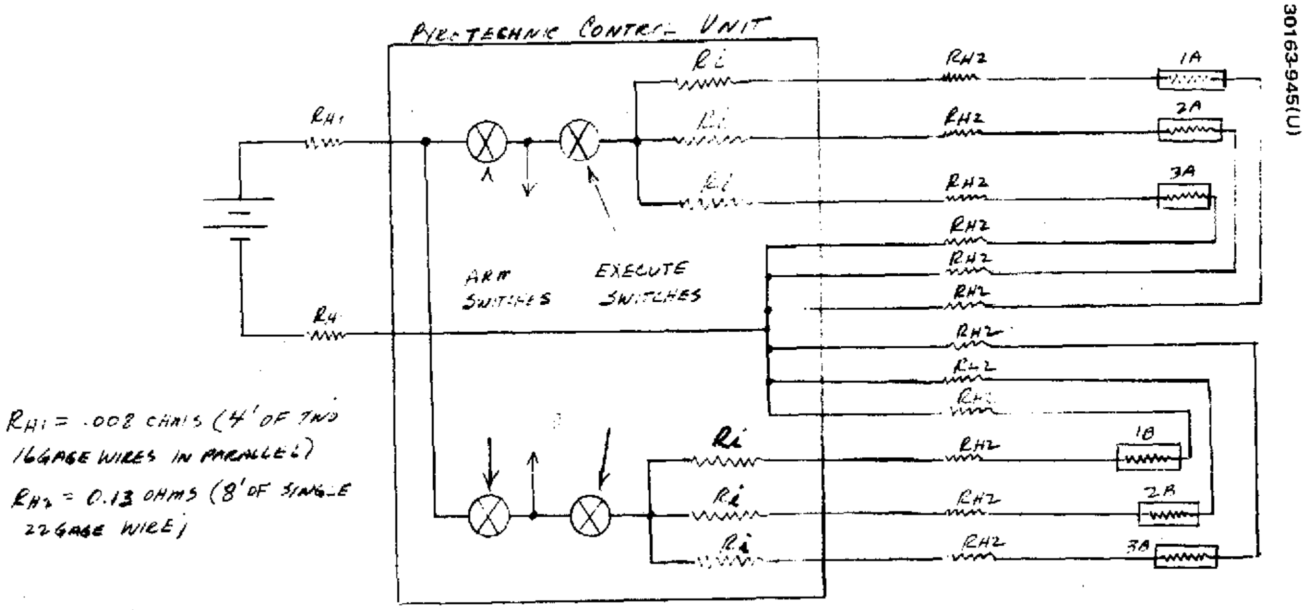
The chief disadvantage is that large capacitors are required. Rapid sequential squib firing (such as large probe chute deployment, in flight disconnect, and aeroshell jettison) requires a dedicated capacitor bank for each individual function because there is not adequate time to recharge one capacitor bank.

Simplified Multiple Squib Driver (Configuration D)

Figure 4-15 represents a simplified version of Figure 4-12. Squibs are grouped together functionally through a common execute switch (through isolation resistors). This configuration reduces the overall number of required power transistors to a reasonable value and still meets all system requirements. This configuration has been selected as baseline.

Redundancy Requirements

Since many pyrotechnic functions are mission critical, it is important that suitable redundancy be provided. Yet, the number of initiators and squib drivers must be held to a reasonable number, especially in the probes where weight and volume are critical. Therefore, the following ground rules have been adopted. All mission critical events will have redundant arm and



	TWO 21-CELL NiCd DIODE ISOLATED BATTERIES			ONE 18-CELL NiCd BATTERY			ONE 13-CELL AgZn BATTERY		
	Ri (OHMS)	CURRENT (AMPS) MIN	CURRENT (AMPS) MAX	Ri (OHMS)	CURRENT (AMPS) MIN	CURRENT (AMPS) MAX	Ri (OHMS)	CURRENT (AMPS) MIN	CURRENT (AMPS) MAX
EACH BRIDGE WIRE	2.0	5.0	8.2	1.6	5.0	8.8	1.2	5.0	9.8
EACH ARM & EXECUTE SWITCH	2.0	15.0	24.6	1.6	15.0	26.5	1.2	15.0	29.4
TOTAL BATTERY	2.0	30.0	49.2	1.6	30.0	53.0	1.2	30.0	59.2

FIGURE 4-16. BATTERY/PCU/BRIDGEWIRE CURRENT LEVELS

execute switches and redundant initiators. All experiments on the orbiter and probe bus will also have complete redundancy. Probe experiments will utilize a single bridgewire and a single driver unless experimenters specifically request bridgewire redundancy. In the latter cases, redundant bridgewires will be activated by single drivers. Probe jettisonable windows, window covers, and probe deployment mechanisms will also be activated by a single driver.

Battery and Pyrotechnic Control Unit Current Requirements

The minimum recommended current for each pyrotechnic initiator is 5 A. This provides a very adequate safety margin and good simultaneity. The isolation resistance R_i for each bridgewire in the PCU must be selected to provide a minimum current of 5 A (see Figure 4-16). The value of R_i must be selected under the following conditions:

1) Minimum battery voltage

2) Maximum harness drop

3) Maximum PCU arm and execute switch (combined) voltage drop of 4 V

If two 21 cell nickel cadmium batteries are used, each cell will provide approximately 1.05 V. Assuming a diode isolation drop of 1.0 V, 21 V are provided to the PCU. R_i must be 2.0 ohms for each bridgewire, to guarantee 5 A into each bridgewire. This results in a total battery current of 5 x 6 = 30 A.

If bridgewire shorting occurs during squib ignition, and also assuming that the arm and execute voltage drops are reduced to a minimum value of 2 V, current through each individual execution switch can increase to a maximum of 8.2 A. Arming switch and battery current will increase, accordingly, to the maximum values shown in the left hand table of Figure 4-16. The battery and PCU must be able to operate at these current levels.

The middle table of Figure 4-16 gives the same type of current data for an 18-cell nickel cadmium battery which provides 18 x 1.05 = 18.9 V to the PCU (Thor/Delta orbiter). This lower battery voltage results in lower isolation resistance values. This is undesirable because the maximum current values are higher when the bridgewires short.

The right hand table of Figure 4-16 provides the same type of data for a 13 cell silver zinc battery (the present baseline probe and probe bus battery configuration). This battery provides a voltage of 13 x 1.3 = 16.9 V and requires an isolation resistance value of 1.2 ohms. This value provides less isolation between the individual initiators resulting in a larger maximum current when the bridgewires short.

The above data indicate that a high battery voltage is desirable because it allows a high isolation resistor value. This, in turn, isolates the PCU and battery and minimizes the effect of bridgewire shorts.

TABLE 4-22. PROBE BUS INITIATOR AND DRIVER REQUIREMENTS

Function	Number of Initiators	Initiator Type	Number of Drivers (Execute)	Minimum Current per Driver, A	Simultaneity Requirement, ms	Mechanism Type
Bicone antenna deploy*	2 **	Bridgewire				Pin puller
Magnetometer boom deploy*	2	Bridgewire	2**	15		Pin puller
Ultraviolet boom deploy*	2	Bridgewire				Pin puller
Large probe IFD*	2	Hot wire	2	5		Connector disengage
Large probe separation*	6	Bridgewire	2	15	4	Separation nuts (3)
Small probe IFD (3)*	6	Hot wire	2	15		Connector disengage
Small probe separation* (3)	6	Bridgewire	2	15	2	Separation nuts (3)
Ion mass spectrometer breakoff hat eject	2 **	Bridgewire	2**	10		Breakoff hat
Neutral mass spectrometer breakoff hat eject	2	Bridgewire				Breakoff hat
Total	30		12***			

*Mission critical event

**Activated simultaneously

***Two arm switches also required

Pyrotechnic Configurations

Initiator and driver characteristics for the probe bus, orbiter, and the probes are summarized in this section, as are the safety requirements.

Probe Bus

Table 4-22 describes the probe bus pyrotechnic configuration. An execute command will release the bicone antenna, magnetometer boom and ultraviolet boom simultaneously, shortly after launch vehicle separation.

The large probe in-flight disconnect (IFD) will break the electrical interface between the probe bus and the probe. The large probe will then be separated by activating three explosive nuts simultaneously with one command. Three matched spring assemblies will then provide a small separation force to the probe.

The three small probe IFDs will break the electrical interfaces with the probe bus and the probes prior to physical separation. A rigid, semicircular arm with a V-cross section encircles each probe just aft of the aeroshell base diameter and preloads it against the bus structure. The arm is hinged at one end and clamped at the other end by a separation nut mounted on the probe bus. An execution command will actuate the separation nuts and a bolt ejector in each nut will provide sufficient impulse to the arm to force it to rotate at sufficient angular velocity and allow each probe to separate cleanly.

Prior to probe bus entry, one execute command will remove the breakoff hats from the ion mass spectrometer and neutral mass spectrometer.

The bridgewires and squib drivers are cross-strapped to provide full redundancy.

Orbiter

Table 4-23 describes the orbiter pyrotechnic configuration. The magnetometer boom will be deployed shortly after launch vehicle separation. The orbit insertion motor pyrogen ignitor is actuated through a safe and arm device. After orbit is achieved, the neutral mass spectrometer and ion mass spectrometer breakoff hats will be removed.

The bridgewires and squib drivers are cross strapped to provide full redundancy.

Large Probe

Table 4-24 describes the large probe pyrotechnic requirements. The PCU is located in the pressure vessel, but most of the pyrotechnic

TABLE 4-23. ORBITER INITIATOR AND DRIVER REQUIREMENTS

Function	Number of Initiators	Initiator Type	Number of Drivers (Execute)	Minimum Current per Driver, A	Simultaneity Requirement, ms	Mechanism Type
Magnetometer boom deploy*	2	Bridgewire	2	5		Pin puller
Orbit insertion motor* igniter	2	Bridgewire	2	5		Pyrogen igniter
Ion mass spectrometer breakoff hat eject	2 **	Bridgewire	2**	10		Breakoff hat
Neutral mass spectrometer breakoff hat eject	2	Bridgewire				Breakoff hat
Total	8		6***			

*Mission critical event

**Activated simultaneously

***Two arm switches also required

TABLE 4-24. LARGE PROBE INITIATOR AND DRIVER REQUIREMENTS

Function	Number of Initiators	Initiator Type	Number of Drivers (Execute)	Minimum Current per Driver, A	Simultaneity Requirement, ms	Mechanism Type
Parachute deploy*	2	Bridgewire	2	5		Mortar
Pressure vessel/ deceleration mode disconnect	2	Bridgewire	2	5		Cable cutter
Aeroshell jettison*	6	Bridgewire	2	15	4	Separation nuts (3)
Parachute jettison*	6	Bridgewire	2	15	4	Separation nuts (3)
Mass spectrometer breakoff hat eject	2	Bridgewire	1	10		Breakoff hat
Mass spectrometer open/close valves	10	Bridgewire	10	5		Valves
Window eject	6	Hot wire	2	15		Jettison spring (6)
Totals	34		21**			

*Mission critical event

**Two arm switches also required

TABLE 4-25. SMALL PROBE INITIATOR AND DRIVER REQUIREMENTS

Function	Number of Initiators	Initiator Type	Number of Drivers (Execute)	Minimum Current per Driver, A	Simultaneity Requirement, ms	Mechanism Type
Despin rockets*	4	Bridgewire	2	10	4	Igniter (2)
Temperature probe deploy	1	Hot wire				Pin puller
Nephelometer window cover eject	1	Bridgewire	1	15		Gas generator and cover release
Pressure transducer inlet eject	1	Bridgewire				Pin puller
Totals	7		3**			

*Mission critical event

**Two arm switches also required

devices are located outside the pressure vessel and must be activated through harness wires that pass through pressurized penetration feed-throughs. The arming switch also serves as a power on/off switch. This switch must turn off power to the PCU completely to prevent use of any battery energy during the long 20 day cruise period.

The first pyrotechnic event during the large probe descent is to eject the parachute by means of a mortar on the deceleration module. This event is initiated by a g-switch on the downward deceleration curve. All subsequent pyrotechnic events are initiated by an internal timer. One sec after chute ejection, the pressure vessel/deceleration module cable cutter is actuated to sever their electrical interconnections. Three sec after chute ejection, the aeroshell is jettisoned by activating three separation nuts simultaneously. Shortly after aeroshell separation, the neutral mass spectrometer breakoff hat will be ejected. Ten pyrotechnic valves will be opened and closed periodically during probe descent to gather and analyze atmospheric samples. Approximately 10 min after aeroshell jettison, the chute will be separated by activating three separation nuts. Experiment external windows will be jettisoned at a height of 40 km.

Small Probe

Table 4-25 describes the small probe pyrotechnic requirements. Once again, the arming switch must turn off all power to the PCU. All pyrotechnic devices are located outside the pressure vessel. Two despin rockets are activated on each small probe shortly before atmosphere entry. The nephelometer window is jettisoned, the temperature sensor is deployed, and the pressure transducer inlet plug is ejected by a single command shortly after atmosphere entry.

Safety Requirements

All harnesses between the PCU and initiators and associated connectors must be properly shielded to prevent premature ignition in high intensity rf fields.

Electrical arming plugs must be provided for the four probes, probe bus, and orbiter to prevent injury to personnel by inadvertent ignition of a pyrotechnic device. These plugs interrupt power between the PCU and the initiators. They are installed during launch preparations.

The orbit insertion motor requires a safe and arm device. In the safe position, a motor actuated mechanical barrier interrupts the explosive power train, disconnects the squibs from the PCU, and shorts the squib bridgewire terminals together. A safety pin provides positive mechanical lock that prevents motor movement from the safe to arm position. The pin is removed shortly before launch vehicle fairing installation. The igniter is armed just prior to liftoff by rotating the mechanical barrier and connecting the squib bridgewires to the PCU. Safing and arming after the safety pin is removed must be controlled from the blockhouse through the umbilical connector. A visual safe and arm indication must be provided at the blockhouse.

4.5 PROGRAMMABLE ON-BOARD DATA PROCESSOR

In the search for mass reduction techniques on the Thor/Delta configuration, the alternative of using a centralized on-board data processor for performing major portions of the spacecraft command and data handling functions was investigated. This can be accomplished using a small general purpose computer on a time-shared basis. The studies show that this approach can yield up to a 26 percent reduction in the mass of the orbiter subsystem equipment, with attendant reductions in power and volume of up to 30 and 80 percent respectively; namely, these savings are up to 5.7 kg (12.6 lb), 5.3 W, and 10,080 cm^3 (615 in^3). Estimates show that this implementation can also result in a significant cost reduction of up to 16 percent, plus an improvement in reliability, for the command and data handling subsystem on the probe bus and orbiter spacecraft. In addition, an on-board processor provides various flexibility and adaptability features that are potentially very useful for optimizing spacecraft operation.

Although a programmable on-board data processor offers these many attractive features, the study conclusion is that this approach is not recommended for the baseline design on this program. The rationale of this conclusion is based primarily upon two of the established evaluation criteria: 1) even though several small computers utilizing LSI technology have been developed, they have not yet been qualified for a spacecraft application and therefore cannot presently be considered to be available space hardware; and 2) being a departure from the more conventional, space proven design methods that employ special purpose electronics, this approach introduces a certain degree of risk that is not felt to be warranted on this program.

As a footnote to this conclusion, it is expected that future space programs will benefit considerably through use of these small general purpose computers for various on-board data handling and control functions. Currently, the major problem with this approach is the lack of space qualified hardware. Eventually, some will be qualified and be available to use for these types of spaceborne applications, probably within the 1974/1975 time frame.

The tradeoff study results are discussed in the following sections. However, these results do not reflect all of the spacecraft functions that were investigated. Initially, this study also considered use of the same centralized on-board data processor to perform major portions of the spacecraft attitude control electronics functions in addition to the command and data handling functions. It was established that the computational and control capability of the central processor was sufficient to perform all three of these spacecraft functions on a time-shared basis. Memory and processing speed estimates indicated that less than 4K words of program and scratch-pad memory were required and less than 50 percent of the machine's computational capacity would be used in order to accommodate and execute the necessary software programs for all three functions. Although this additional time-sharing of equipment would result in even greater mass, volume, and power savings, it was decided early in the study to concentrate only on the command and data handling functions in order to simplify the centralized equipment and analysis. Therefore, even though attitude control functions are considered to be a prime candidate for an on-board processor implementation, this application is not included in the following tradeoff discussions.

Centralized Command and Telemetry Data Handling Subsystem

Figure 4-17 is a simplified block diagram showing the basic concepts of a centralized command and telemetry data handling subsystem that was considered for the Thor/Delta configuration of the probe bus and orbiter spacecraft. The centralized subsystem consists of seven basic types of units divided into three functional equipment categories as follows:

1) Time-shared central computer equipment
    a) Dual processor and I/O
    b) Dual memory
2) Command equipment
    a) Dual command demodulator and configuration selector
    b) Remote decoders (six required)
    c) Dual pyro control unit
3) Telemetry equipment
    a) Dual PCM encoder
    b) Remote multiplexers (seven required)

The time-shared data processor and I/O combination, plus the memory, form the central elements in this configuration. These elements are implemented by use of a small general purpose computer that performs a relatively large portion of the command and telemetry functions. Power supplies are not shown in Figure 4-17 in order to simplify the diagram. The shaded areas in the figure represent fully redundant, dual units. Any combination of these dual units can be used to perform the necessary

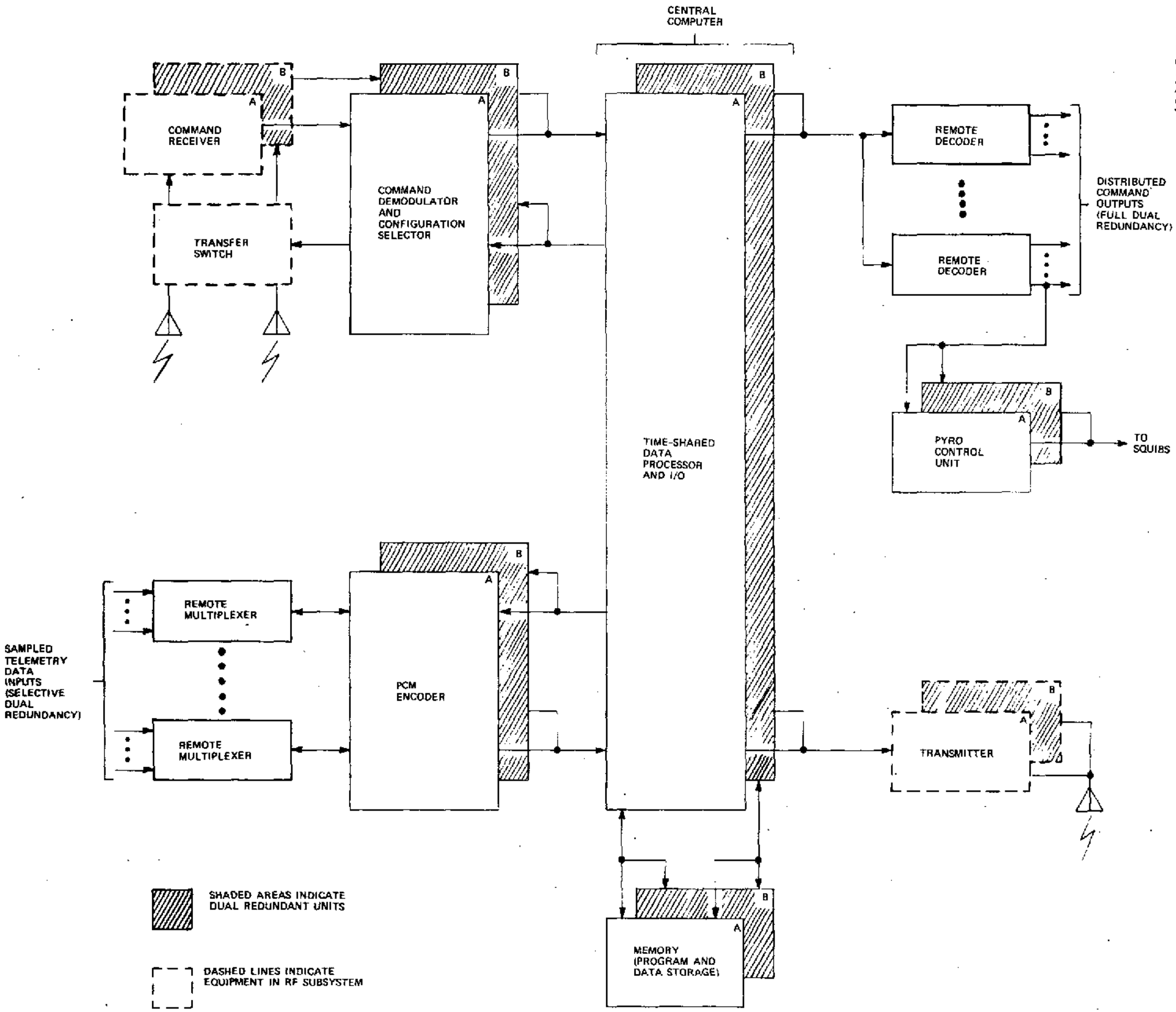


FIGURE 4-17. CENTRALIZED COMMAND AND TELEMETRY DATA HANDLING SUBSYSTEM

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functions. Redundancy is achieved in the remote multiplexers and the remote decoders through the redundant assignment of data input and command output channels. The PCM encoder, remote multiplexers, and remote decoders are equipment items that were previously developed by Hughes for the OSO spacecraft program. For the sake of completeness some of the rf system interfacing equipment is also shown in Figure 4-17. This interfacing equipment is shown in dashed lines; namely, the command receiver, the antenna/receiver transfer switch, the transmitter, and the associated antennas.

Table 4-26 presents a summary of estimated mass, volume, and power characteristics of the centralized data subsystem that is depicted in Figure 4-17. The table includes estimates of the subsystem configurations for both the probe bus and orbiter spacecraft.

For the purpose of this tradeoff study, the characteristics of the CDC-469 computer were assumed. The CDC-469 was selected for this tradeoff analysis primarily because of its small physical size, low power, and current production status. The basic characteristics of this computer, plus the characteristics of several other candidate computers, are shown in Table 4-28 in a later subsection.

A comparison of the centralized subsystem characteristics versus separate, non-centralized command and data handling subsystems is shown in Table 4-27. This table summarizes and compares the mass, volume, and power characteristics of both of these subsystem design approaches for both the probe bus and orbiter spacecraft. In addition, the table shows a comparison of the dual central computer characteristics with the three particular units of the non-centralized subsystems that it replaces; namely, a dual central command decoder, a dual telemetry processor, and a data storage unit. These tradeoff estimates indicate that significant reductions in mass, volume, and power are possible through the use of a centralized on-board processor, especially on the orbiter spacecraft where 393 kilobits of data storage is required.

Reliability of the centralized subsystem is predicted to be somewhat better than for the separate, non-centralized subsystems approach. The reliability aspects of the hardware and software are discussed at length in a later subsection.

The basic functions of each of the units shown in Figure 4-17 are briefly described in the following paragraphs.

TABLE 4-26. CENTRALIZED COMMAND AND TELEMETRY DATA HANDLING SUBSYSTEM – CHARACTERISTICS SUMMARY

Equipment Description	Probe Bus					Orbiter				
	Mass (weight)		Volume		Power W	Mass (weight)		Volume		Power W
	kg	(lb)	cm^3	(in^3)		kg	(lb)	cm^3	(in^3)	
Dual central computer	5.1	(11.2)	4,590	(280)	5.0	7.3	(16.0)	5,900	(360)	6.0
Dual command demodulator/ configuration selector	2.1	(4.6)	3,080	(188)	3.0	2.1	(4.6)	3,080	(188)	3.0
Remote decoders (6)	1.8	(4.0)	1,720	(105)	0.3	1.8	(4.0)	1,720	(105)	0.3
Dual pyro control unit	1.4	(3.0)	1,670	(102)	---	0.9	(2.0)	1,130	(69)	---
Dual PCM encoder	2.8	(6.3)	3,500	(214)	2.6	2.8	(6.3)	3,500	(214)	2.6
Remote multiplexers (7)	1.4	(3.0)	970	(59)	0.3	1.4	(3.0)	970	(59)	0.3
Subsystem Totals	14.6	(32.1)	15,530	(948)	11.2	16.3	(35.9)	16,300	(995)	12.2

Note: Each of the dual units contain the necessary cross-strap switching and dual power supplies.

TABLE 4-27. CENTRALIZED VERSUS NONCENTRALIZED SUBSYSTEMS – COMPARISON SUMMARY

Equipment Description	Probe Bus Mass (weight) kg	(lb)	Probe Bus Volume cm^3	(in^3)	Probe Bus Power W	Orbiter Mass (weight) kg	(lb)	Orbiter Volume cm^3	(in^3)	Orbiter Power W
Noncentralized Equipment Replaced by Central Computer										
Command subsystem: Dual central decoder	4.3	(9.4)	6,000	(366)	3.6	4.3	(9.4)	6,000	(366)	3.6
Data handling subsystem: Dual telemetry processor	3.2	(7.2)	4,670	(285)	5.7	3.2	(7.2)	4,670	(285)	5.7
Data storage unit	---		---		---	5.5	(12.0)	5,310	(324)	2.0
Equipment Comparisons										
Equipment replaced by computer	7.5	(16.6)	10,670	(651)	9.3	13.0	(28.6)	15,980	(975)	11.3
Dual central computer	5.1	(11.2)	4.590	(280)	5.0	7.3	(16.0)	5,900	(360)	6.0
Noncentralized subsystems, total	17.0	(37.5)	21,610	(1319)	15.5	22.0	(48.5)	26,380	(1610)	17.5
Centralized subsystem, total	14.6	(32.1)	15,530	(948)	11.2	16.3	(35.9)	16,300	(995)	12.2
Estimated computer savings	2.4	(5.4)	6,080	(371)	4.3	5.7	(12.6)	10,080	(615)	5.3
Percent reduction of subsystem	14 percent		28 percent		27 percent	26 percent		38 percent		30 percent

Notes:

1) Each of the dual units contain the necessary cross-strap switching and dual power supplies.

2) The dual central computer consists of two combined processor and special I/O modules, two power supplies, cross-strap switching, and memory modules as follows: two 4K modules for the probe bus and two 16K modules (or four 8K modules) for the orbiter.

Time-Shared Central Computer

The central computer equipment consists of two basic types of modules; namely, the combined processor and I/O modules and the memory modules. Two of each of these module types are used to provide complete dual redundancy. Either processor-I/O module can be operated with either or both memory modules.

On the probe bus, two 4 kilobit x 16 bit memory modules are used, while two 16 kilobit memory modules (or four 8 kilobit modules) are used on the orbiter spacecraft to provide additional data storage capability. Software estimates indicate that less than 2 kilobit words are required for the basic program storage, including program alterable scratchpad/data memory, for either spacecraft configuration. However, 4 kilobits of program memory are provided inorder to allow a reasonable margin for growth. The portions of memory that are reserved for program storage can be protected, in 1024 word increments, by means of "disable write" circuitry. This effectively allows the program to reside in a read only portion of memory to protect it from being altered inadvertently. However, this memory protection feature can be over-ridden, by means of a proper sequence of commands from the ground, if reloading or altering of the program is desired.

The memory capacity of two 16 kilobit modules on the orbiter spacecraft can be allocated in the following manner:

1) 4 kilobits of each module for program and scratchpad usage.

2) 12 kilobits of each module for data storage.

This provides a total data storage capability of 24,576 words of 16 bits each for a total of 393,216 bits.

The time-shared functions of the central computer are estimated to require less than 25 percent of the machines computational capacity. These functions are briefly described in later subsections.

Reliability. A rigorous reliability analysis of the computer and centralized subsystem was not performed during this study. However, the reliability of this subsystem is expected to be somewhat better than that of the comparable, non-centralized command and data handling subsystems. This expected improvement is based upon a number of reasons that would tend to support such a prediction. Some of these are:

1) The centralized subsystem contains less electronic component parts; this is evidenced by the lower mass. It is due to a high usage of LSI technology in the computer and also to the fact that the computer electronics are time-shared to a high degree for

performance of many functions. Conventional reliability analysis and prediction is highly influenced by the number (and types) of electronic components.

2) The dual computer assembly has an additional level of cross-strapped redundancy that is not contained in a special purpose electronics configuration. That is, the memory modules are cross-strapped to operate with either of the processor-I/O modules; these two types of modules are used in lieu of a single, conventional special purpose electronics unit. Modularization of equipment, so as to provide additional cross-strapping points between redundant modules, tends to improve reliability appreciably.

3) Reliability (and performance) can be enhanced by the capability to reprogram the system while in space. This feature can allow more work-around methods and possibilities of compensating for a malfunction in the event a failure occurs (or of compensating for slight deviations from predicted performance after some space operational experience has uncovered any incipient design deficiencies).

4) Several other spaceborne computer applications studies performed by Hughes have indicated that reliability is definitely enhanced by this type of a subsystem implementation (e.g., studies related to the JPL Outer Planets Mission program and to military space programs).

The contribution of software to the reliability of a centralized subsystem is a relatively intangible element. Good theoretical tools have not yet been developed for including software affects in the equations for analysis of system reliability. However, even though the tools for providing a high confidence measurement and prediction of software reliability are not available, the software can be assumed to be 100 percent reliable, especially when precautions are taken to compensate for this unknown factor. That is, if software has any affect on the reliability of a system, its influence can be cancelled out by providing the capability to dump the memory of the computer (to allow analysis on the ground) and to reload new programs into memory while the spacecraft is in space.

Prediction of perfect, 100 percent software reliability can be rationalized when the following type of argument is used. Namely, the system will perform <u>exactly</u> as directed by the software program. Any performance glitches must be attributed to one of the following reasons:

1) Software design error — The possibility of design errors is not normally taken into consideration in conventional reliability analysis and prediction techniques. The detection and correction

of design errors is normally expected to be accomplished prior to launch, since this is one of the objectives of a comprehensive spacecraft test program. If design errors remain undetected, the testing was evidently not sufficient; these incipient design deficiencies should not be attributed to erroneous reliability analysis and prediction. (Although, the reliability analysis procedures are really inadequate in that they cannot account for these eventualities.)

2) Hardware failure.

3) Software program disturbance induced by a hardware "glitch" or failure.

Using this type of rationale, no detrimental reliability affects can be attributed to software. System performance problems can only be attributed to inadequate testing or to hardware. It appears that software has no affect on reliability.

Command Functions

Four types of units are involved with the handling of spacecraft commands in this centralized subsystem. The functions of each of these types of equipment modules are briefly described below.

Command Demodulator/Configuration Selector. Upon receipt of the receiver in-lock indication and subcarrier signals from the command receiver located in the rf subsystem, the proper command demodulator/configuration selector unit is activated, by recognizing the proper addressable frequency, and it begins operation. The unit remains activated until loss of the subcarrier, at which time it is shut down. The real time uplink PSK subcarrier is received and demodulated into digital command data and clock signals. These incoming data and clock signals are then transmitted to the central computer for processing of the commands.

In addition to its primary function of command demodulation, important equipment selection functions are also performed by this unit. It controls activation of the desired set of redundant computer modules as selected by the ground. This is accomplished by means of several unique commands that are recognized and decoded by the configuration selector portion of the unit. As determined by these decoded commands, on/off control signals are generated to activate a particular processor-I/O module and memory module combination that is selected for operation. The execution of these configuration commands is completely independent of, and cannot be affected by, the central computer itself. Once activated, the selected configuration of computer modules will remain in operation until changed by ground command. After the selected computer modules have been activated in this manner, the computer receives command instructions from the ground to activate the desired combination of all other redundant units in the subsystem. Upon receipt of these instructions, the computer subsequently issues specific commands that control the necessary on/off switching to activate the desired units. This total configuration selection sequence allows the ground to have full control for selecting any combination of redundant units within the subsystem.

Another important equipment selection function of the command demodulator/configuration selector unit is controlling the state of the antenna/receiver transfer switch in the rf subsystem. This switch determines the connection combination of the two omniantennas with the two command receivers. In the event the spacecraft has not received valid command data within a predetermined number of hours, a timer in this unit generates an antenna reverse pulse. When valid commands have been received, a polycode verification check by the central computer normally resets this timer in order to maintain the current antenna/receiver connections. The timer can also be reset by real-time or stored commands issued by the computer. Likewise, an antenna reverse pulse can be forced by real-time or stored commands issued from the computer if it is desired to reverse the antenna/receiver connections.

Central Computer. The selected computer receives and accumulates the uplink command data and clock from the command demodulator/ configuration selector unit. Computer processing of the received data then consists of a number of basic command management functions including:

1) Command decoding, polynomial code check for command verification, command reformatting, and encoding for distribution.

2) Command distribution as appropriate. Real time commands, requiring immediate execution, are manchester encoded and transmitted directly to the remote decoders. Stored commands, that are to be executed at a later time, are placed in the stored command memory.

3) Storage of time dependent commands. Event dependent commands may also be accommodated if required. These stored commands consist of pulsed commands, 16 bit serial magnitude commands, and time-delay commands that determine when the pulse and magnitude commands are to be executed.

4) Processing, control, and timing of the various types of stored commands to ensure their initiation and distribution to the remote decoders at the proper time.

Remote Decoder. The remote decoders receive command addresses and manchester encoded command data from the central computer. The appropriate decoder recognizes and decodes the address, decodes the command data into pulse or magnitude commands, and routes these commands to the proper user on the spacecraft. The distribution of the commands to the subsystem and experiment users is fully redundant; i.e., the same commands can be issued by either of the two independent remote decoders.

Pyro Control Unit. This unit contains the necessary high current squib drivers for firing the pyrotechnic devices on the spacecraft. The squib drivers are enabled by receipt of two unique, contiguous pulse commands from a remote decoder.

Telemetry Data Handling Functions

In this centralized subsystem, three different types of units are involved with spacecraft telemetry and data handling operations. The functions of each of these types of equipment modules are briefly described below.

Central Computer. This unit is the central element for the control and management of telemetry and data handling functions. These basic control and processing functions include:

1) Telemetry format control — This includes generation of basic control signals for synchronization and assembly of telemetry data into desired data formats. Several types of control functions are performed, such as:

   a) Generation of addresses of the signal data to be sampled, in prescribed sequences, for insertion into the real-time downlink telemetry data stream or for storing into data storage memory for later transmission. Multiple telemetry data formats are generated to accommodate the various data transmission and data storage requirements at prescribed times during the mission scenario. This signal address data is manchester encoded and serially transmitted to the PCM encoder unit.

   b) Generation of frame synchronization and identification codes.

2) Telemetry timing control — This includes generation of basic timing signals to control the major telemetry functions; i.e., gathering, storing, and transmission of telemetry data. These timing signals assure the synchronization of data flow and select the proper rates for data manipulation, such as:

   a) Data sampling rates of both the remote multiplexers and of the stored data to be retrieved from memory.

   b) Transmission bit rates to accommodate communication link constraints during various phases of the mission.

3) Telemetry data routing — Pulse code modulated NRZ-L data are received serially from the PCM encoder unit. Further management and distribution of the data then commences as required. This primarily consists of routing the data to the appropriate

destination, depending upon whether the data is intended for storage or for real-time telemetry, i.e., to/from data storage or to processing functions that condition the data before transferring it to the rf subsystem for transmission.

4) Telemetry data processing — This includes control and execution of the various digital processing algorithms that are necessary to prepare the data for transmission, e.g., functions such as convolutional encoding, modulation index selection, and biphase modulation.

5) Telemetry data storage — This includes memory storage/retrieval capability, for both data and appropriate timing control information, when it is desired to save the data for transmission at a later time.

PCM Encoder. This unit receives serial manchester encoded address data from the central computer. The address(es) indicate the desired data source(s) to be sampled. Part of the address word is first decoded, to determine which remote multiplexer to interrogate, and then the address is relayed to the proper multiplexer. After the multiplexer has sampled the proper channel, the requested data are received back from the remote unit on a serial line and in burst PAM form. These data are then encoded into a PCM NRZ-L format. When the incoming data represent the sampling of an analog signal, it is converted into an appropriate 8 bit PCM digital word. This unit then serially transmits the PCM encoded data back to the central computer for further processing.

Remote Multiplexer. These modules receive the channel select portion of the signal address from the PCM encoder unit in manchester encoded form. They then sample the requested data point and transfer the signal information back to the PCM encoder. Each of these remote units has 32 input channels for gathering telemetry data of various types; namely, high level analog data, parallel bilevel digital data, and serial digital data. During the telemetry data gathering process, these remote units commutate the various types of data samples onto a PAM output line for multiplexed transmission back to the PCM encoder unit.

Computer Hardware Survey

A relatively extensive survey of aerospace computer developments was conducted in order to find appropriate computer hardware that might be available for Pioneer Venus application. The search concentrated on finding small general purpose computers with very low mass and power characteristics, e.g., less than 4.5 kg (10 lb) and 10 W with 4K words of memory. Although the primary objective was to locate small machines that were already fully developed, many of the computers that were reviewed were either still in the development stage or were larger and more power consuming than desired.

TABLE 4-28. CHARACTERISTICS OF SOME REPRESENTATIVE CANDIDATE COMPUTERS

Manufacturer and Type	Physical Characteristics						Memory Features		
	Mass (Weight)		Volume		Power, W	Component Technology	Word Length, bits	Capacity, words	Type
	kg	(lb)	cm^3	(in^3)					
CDC-469	1.8	(4)	1,165	(70)	4	P-MOS LSI	16 (32)	8K (8K to 32K)	Plated wire (core, semi)
Autonetics D-216	2.7	(6)	4,100	(250)	32	P-MOS LSI	16 (24, 32)	8K (2K to 64K)	Plated wire
GSFC AOP	3.7	(8)	3,280	(200)	5 to 10	TTL LSI (CMMA)	18 (24, 33)	4K (4K to 64K)	Plated wire (core, semi)
Honeywell HDC-301	1.4	(3)	1,300	(80)	23	P-MOS LSI	16	4K (2K to 32K)	Semiconductor (core, plated wire)

Manufacturer and Type	Other Computer Features					
	Double Precision		Add/Multiply/ Divide Time, µsec	Number of Instructions	Number of Interrupts	Comments
	Add/ Subt.	Mult./ Div.				
CDC-469	H	S	6-8/26/76	42	3	Optional clock speeds of 1, 2, 2.5 Mhz; characteristics are for 1 MHz clock; also has a "direct execute" interrupt
Autonetics D-216	H	H	2.5/13.75/23.75	68	1	Characteristics do not include case
GSFC AOP	S	S	4/30/60	55	16	Memories are power switched and dissipation depends on access rate; includes volume reserved for I/O control and interface buffers
Honeywell HDC-301	H	S	5/21/65	47	1	Characteristics include 4 PC cards only without external case; 1 MHz clock; also has a "power recovery" interrupt

NOTES:

1) Characteristics do not include power supplies; they include processor, I/O, and memory only.

2) Characteristics are based on the word length, memory capacity, and memory type shown (optional word lengths, range of memory capacity, and memory types are shown in parenthesis).

3) Instruction execution times shown are for single precision.

4) Data flow is parallel for all machines.

5) Double precision notations are H for hardware implementation and S for software implementation.

A general conclusion of the survey is that there are only a few computers of the desired class that have been fully developed. Although a few of these machines can be considered as available, none of them have yet been qualified for use on a space program. Based upon this lack of space proven or space qualified hardware, none of the machines have been recommended for Pioneer Venus application.

Table 4-28 gives a summary of some of the basic characteristics for four of the candidate computers that were investigated. These four machines are considered to be some of the best candidates for Pioneer Venus application due to their development status and their size and power characteristics. However, it is not intended to infer that these are the only aerospace computers that were considered as potential candidates for this space application during the course of this investigation. Rather, the table is intended to show only some of the representative characteristics of the class of computers that are of interest to Pioneer Venus.

Some of the computers that were investigated merit some discussion because of their advanced development status. There is one general purpose computer of relatively small size that has recently been employed on a space program. However, its mass and power characteristics are slightly higher than could be justified for a Pioneer Venus application. This machine is the on-board processor (OBP) developed by NASA Goddard Space Flight Center. With 4K of core memory, the machine weighs approximately 8 kg (18 lb) and dissipates approximately 28 W excluding the power supply. The on-board processor is currently being used to perform a number of telemetry, command, and control functions as an experiment on the OAO-C spacecraft that was launched in August 1972. The functional characteristics and features of this machine are considered to be well suited for performing these types of functions on a spacecraft, if only the mass and power were lower.

In recognition of the need for a computer with lower mass and power on future spacecraft programs, GSFC is presently well along with the development of an advanced LSI version of the OBP. It is called the advanced on-board processor (AOP) and its characteristics are summarized in Table 4-28. The advanced on-board processor hardware and software are currently in an advanced stage of development, and the potential for many future applications to various types of spacecraft functions is foreseen. Development activity on the advanced on-board processor has concentrated on its use as a centralized command and telemetry data processor. As such, it is designed to interface with the remote command decoders and remote telemetry multiplexers that were developed for the OSO program by Hughes. This combination of equipment provides a very modular data handling and control system that can be configured to meet the specific requirements of a spacecraft. Several flight models of the AOP are planned to be built in the near future. The advanced on-board processor is planned to be used on the ERTS-B and International Ultraviolet Explorer (IUE) programs. It is being considered for use on a number of additional programs including Nimbus G, EOS, OSO-J, and SATS. The first planned application of the AOP was for the High Energy Cosmic Ray experiment on the HEAO-A spacecraft, but completion of the HEAO program has been delayed.

The CDC-469 appears to be the only candidate computer that is in relatively large production at the present time. It is also one of the smallest in physical size and lowest in power, making it one of the most attractive candidates. In addition, a large amount of supporting software has already been developed and debugged. It has been designed and tested to meet stringent Military Specifications and has already been considered for use on a number of space programs. This machine is considered to be a major contender for space usage, but it has not yet been qualified for a space application.

The Autonetics D216 is a 16-bit member of the D200 family of computers. This series of LSI computers was designed to provide capability for a wide spectrum of military and space applications. Besides a number of missile and aircraft avionics applications, it is being considered for use on a number of space missions including the Space Shuttle and unmanned satellites. A D216S computer is currently planned to be employed in a Space Test Program satellite experiment being built by the Space Division of Rockwell International for SAMSO. This is the first D216S to be provided by the Autonetics Division for space flight. The satellite, which will carry this experiment plus three other experiments to gather scientific data, is planned to be launched in early 1974.

The HDC-301 is a small LSI computer that is currently under development at Honeywell. The computer is designed to use semiconductor memory modules although, like a large number of the other computers that were investigated, it can optionally accommodate either plated wire or core memory modules. The machine has primarily been designed for use in missile and aircraft avionics systems, and is presently planned to be employed in several of these types of applications. Its small size makes it a potential candidate for space usage, although no plans for the space application of this machine are presently known.

Other computers that were investigated as potential candidates include:

- Several other versions of the CDC-469 computer that are currently in development by CDC.
- Other Rockwell International (Autonetics Division) computers of the D200 series class.
- The Viking Lander 1975 computer being developed by Martin Marietta and Honeywell for NASA Langley Research Center.
- The Viking Orbiter 1975 computer being developed by the Jet Propulsion Laboratory.
- The HDP series of computer developments by Hughes.

- The programmable spacecraft control processor development for the International Telecommunications Satellite Consortium (Intelsat) and COMSAT by the TRW Systems Group.

- The SKC-3000 LSI computer development by the Kearfott Division of the Singer Company.

- European spaceborne computer developments by Selinia and by CNES for the ESRO.

- Plus a number of other aerospace computer developments by the NASA agencies, SAMSO, AC Electronics, Arma, Bendix, Bunker Ramo, Burroughs, IBM, Litton, Nortronics, Raytheon, RCA, Rohm, Sperry Rand, Teledyne, TI, UNIVAC, and others.

REFERENCES

4-1 "Probe Bus/Orbiter Spacecraft and Probes Command Interface Design," Hughes Aircraft Co., HS 507-0022-148, Study Task No. CC6, dated 23 April 1973.

4-2 "Probe Bus/Orbiter Spacecraft and Probes Data Handling Interface Design," Hughes Aircraft Co., HS 507-0022-149, Study Task No. DH 4, dated 23 April 1973.

4-3 "Analyze Prevention of Inadvertent Irreversible Command Execution," Hughes Aircraft Co., HS 507-0022-68, Study Task No. CC8, dated 4 January 1973.

4-4 "Probe Data Storage," Hughes Aircraft Co., HS 507-0022-38, Study Task No. DH5, dated 15 November 1972.

4-5 "Probe Data Rate," Hughes Aircraft Co., HS 507-0022-37, Study Task No. DH6, dated 15 November 1972.

5. THOR/DELTA BASELINE DESCRIPTION

This section briefly describes the baseline designs for the Thor/Delta configuration of the command and telemetry data handling subsystems for the four space vehicles; namely, for the probe bus and orbiter spacecraft and for the large and small probes. These subsystem designs were established prior to the decision to use the Atlas/Centaur launch vehicle; they therefore do not represent the current baseline and are briefly documented here for information purposes only. The current Atlas/Centaur baseline is described in Section 6, where the subsystems are discussed in more detail.

The Thor/Delta designs described here employ custom large scale integration (LSI) circuit technology to a large degree in order to minimize the mass of the subsystems. A summary of mass, power, and volume characteristics is shown in Table 5-1; it includes the command and data handling subsystems for all four vehicles.

Subsection 5.1 describes the command and data handling subsystems for both the probe bus and orbiter spacecraft, and the commonality and differences are pointed out. The subsystems are essentially identical, and employ a large amount of existing OSO hardware. The major difference is the addition of a data storage unit on the orbiter spacecraft, with minor differences in the pyro control units and in the telemetry data formats.

The command and data handling subsystems for the large and small probes are described in Subsection 5.2. These subsystems are also common to a high degree. The small probe subsystem is essentially a subset of the large probe, since it is functionally similar, but requires fewer squib drivers and less circuitry for data gathering and commands. Besides the large amount of functional and circuit commonality between the probes, there is also some additional functional and circuit commonality between the probes and the spacecraft, particularly in the area of user interfaces. The hardware for the probes has limited physical commonality due to the stringent packaging constraints.

5.1 SPACECRAFT COMMAND AND DATA HANDLING SUBSYSTEMS

The command and data handling subsystems are highly modular in nature and are essentially identical on the probe bus and orbiter spacecraft. The few differences that exist are explained in the following discussions.

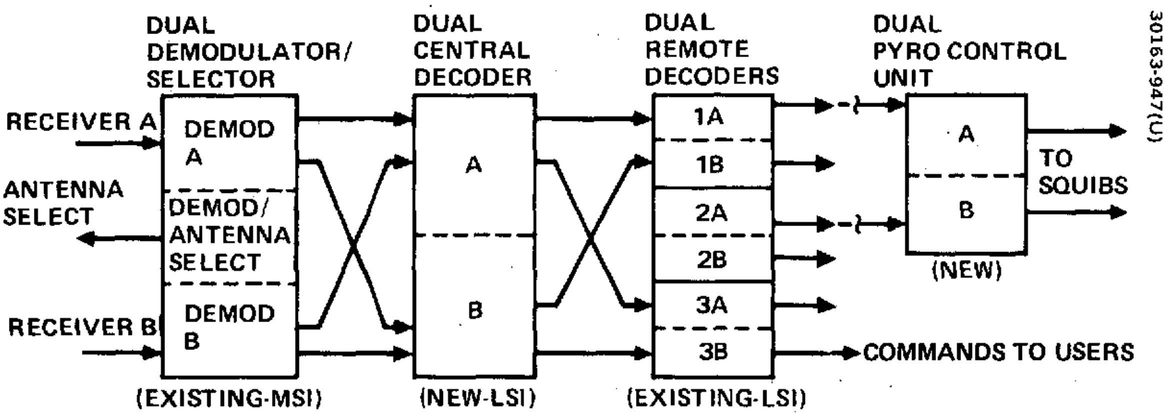


FIGURE 5-1. MODULAR COMMAND SUBSYSTEM

TABLE 5-1. CHARACTERISTICS SUMMARY - COMMAND AND DATA HANDLING SUBSYSTEMS

Thor/Delta Vehicle Subsystems	Characteristics: Mass (Weight) kg	(lb)	Power, W	Volume, cm^3	(in^3)
Probe bus					
Command subsystem	7.1	(15.5)	6.9	9,600	(586)
Data handling subsystem	5.8	(12.8)	8.6	7,060	(431)
Totals	12.9	(28.3)	15.5	16,660	(1017)
Orbiter					
Command subsystem	6.6	(14.5)	6.9	9,060	(553)
Data handling subsystem	9.9	(21.8)	8.9	9,350	(571)
Totals	16.5	(36.3)	15.8	18,410	(1124)
Large probe			Cruise/descent		
Command and data handling subsystem	2.1	(4.7)	0.06/4.2	2,620	(160)
Small probe					
Command and data handling subsystem	1.2	(2.7)	0.06/3.1	1,340	(82)

Command Subsystem

Figure 5-1 shows a block diagram of the command subsystem. It consists of four basic types of units, each of which employs complete dual redundancy. They are the dual demodulator/selector, the dual central decoder, three dual remote decoders, and the dual pyro control unit. The following is a brief description of the major functions performed by each of these units.

1) Dual demodulator/selector

- Receives and demodulates subcarrier into command data and clock signals
- Provides receiver reverse switching to select the connection combination between the receivers and antennas

2) Dual central decoder

- Decodes and processes the incoming command data
- Performs a polynomial code check for command verification
- Provides storage for time dependent command sequences
- Initiates execution of the commands at the proper time
- Distributes commands to the proper remote decoder

3) Dual remote decoder

- Recognizes and decodes the appropriate command addresses
- Receives and decodes command data into pulse or magnitude commands
- Distributes pulse commands and magnitude data to spacecraft users (and to probes on probe bus)

4) Dual pyro control unit

- Receives interlocked pulse commands from remote decoders
- Provides high current switching to fire pyrotechnic devices

Table 5-2 summarizes the basic functional characteristics of the command subsystem for both the probe bus and orbiter spacecraft. As indicated in the table, the characteristics are identical with one exception; the probe bus has 50 percent more pyrotechnic drivers and initiators than the orbiter, since the probe bus requires more squib firing events.

TABLE 5-2. PROBE BUS/ORBITER SPACECRAFT COMMAND SUBSYSTEM FUNCTIONAL CHARACTERISTICS SUMMARY

Parameter	Characteristics	
	Probe Bus	Orbiter
Command mode	Real-time or stored	
Command type	Pulse or magnitude	
Command initiation	Ground command, timer, or event	
Command uplink		
Command word size	36 bits (includes 7 bit error code)	
Bit rate	1 bps	
Modulation	PSK	
Probability of false command	1×10^{-9}	
Number of pulse command outputs	192	
Number of magnitude command outputs	12	
Word size of magnitude commands	10 bits	
Command storage capacity	2048 bits (85 words maximum)	
Resolution of command executions	±125 ms (stored mode)	
Number of pyrotechnic drivers	6 redundant pairs	4 redundant pairs
Maximum delay between concurrent fire pulses	0.1 ms	
Number of pyrotechnic initiators fired (capability)	30	18

Hardware Design Description

A summary of the hardware derivation and characteristics is shown in Table 5-3. In addition to presenting the mass, power, and volume characteristics for both the probe bus and orbiter spacecraft, the table briefly describes the source and type of hardware employed.

TABLE 5-3. MODULAR COMMAND SUBSYSTEM – HARDWARE DERIVATION AND CHARACTERISTICS

Unit	Hardware Derivation		Hardware Characteristics*		
	Thor/Delta Probe Bus	Thor/Delta Orbiter	Mass (Weight), kg (lb)	Power, W	Volume, cm^3 (in^3)
Dual demodulator/ antenna selector	Motorola - POV (Uses 98 percent Viking orbiter circuits; MSI; printed circuit boards)	Same	2.1 (4.6)	3.0	3,080 (188)
Dual central decoder	New design (Uses 50 percent OSO circuits and new LSI; MICAM** and printed circuit boards)	Same	1.8 (3.9)	3.6	3,130 (191)
Dual remote decoder	OSO (three dual units; existing LSI; printed circuit boards)	Same	1.8 (4.0)	0.3	1,720 (105)
Dual pyro control unit	New design (three slice unit; modules)	Same Design (two slice unit)	1.4 (3.0) 0.9 (2.0)	-0- -0-	1,670 (102) 1,130 (69)
		Probe bus totals	7.1 (15.5)	6.9	9,600 (586)
		Orbiter totals	6.6 (14.5)	6.9	9,060 (553)

*Where two sets of values are given, the bottom line is for the orbiter spacecraft.

**MICAM is the acronym for Hughes "microelectronic assembly method"

As indicated in the table, three of the units are planned to be built by Hughes and utilize OSO designs. Of these units, the dual central decoder and dual remote decoder employ custom LSI circuit technology as a means for reducing mass; the pyro control unit does not employ LSI, since this technology is not as amenable to the high current pyrotechnic circuit applications.

The dual demodulator/antenna selector unit is planned to be purchased from Motorola. This unit is a modification of the command demodulator that was developed for the Viking Orbiter program; it utilizes MSI circuit technology.

Data Handling Subsystem

A block diagram of the data handling subsystem is shown in Figure 5-2. It also consists of four basic types of units; remote multiplexers (seven required), dual PCM encoder, dual telemetry processor, and orbiter data storage unit that is required only on the orbiter spacecraft. As implied by the unit names, the PCM encoder and telemetry processor employ complete dual redundancy. Redundancy is achieved in the remote multiplexers through redundant channel assignments of critical data inputs. While the baseline employs a nonredundant orbiter data storage unit, the splitting of the storage into two memories is strongly being considered; this would provide graceful degradation such that, if one fails, 50 percent of the total capacity is still usable.

The following is a brief description of the major functions performed by each of these units:

1) Remote Multiplexer

- Accepts data from spacecraft subsystems and science instruments (and from probes on the probe bus)
- Multiplexes analog, serial digital, and bilevel discrete input data onto a single output line to be transferred to the PCM encoder
- Receives address control data from the PCM encoder to provide channel selection
- Provides signals to data handling subsystem users to control the sampling of digital data

2) Dual PCM Encoder

- Accepts PAM analog data and serial digital telemetry data from remote multiplexers
- Digitizes analog data into 8 bit words and combines it with digital data into a PCM bit stream to be transferred to the telemetry processor

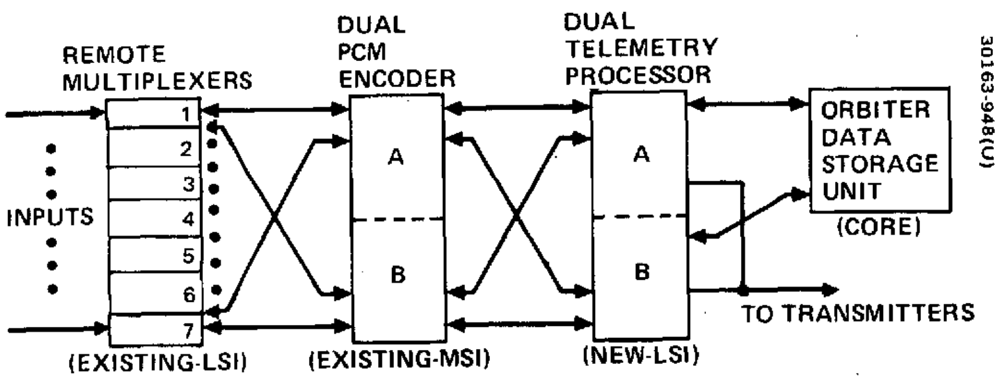


FIGURE 5-2. MODULAR DATA HANDLING SUBSYSTEM

- Relays address control data from the telemetry processor to the remote multiplexers for channel selection

3) Dual Telemetry Processor

   - Provides a versatile telemetry frame format that is based upon preprogrammed sampling sequences and is structured in flight by ground command
   - Generates addresses of data inputs to be sampled
   - Encodes, modulates, and controls amplitude of PCM data and transfers it to the spacecraft transmitters or memory

4) Orbiter Data Storage Unit

   - Provides storage for scientific and engineering data

Table 5-4 presents a summary of the basic functional characteristics for the data handling subsystem. It shows the similarity and differences between the probe bus and orbiter spacecraft. As indicated in the table, the characteristics are identical with two exceptions. First, data storage is only required on the orbiter, and second, the subframe data formats are unique on each of the spacecraft. The hardware is essentially the same for the data formats, however, since they both employ the same number of read-only memory (ROM) devices; the difference is that the ROMs are preprogrammed to provide different formats that are consistent with the different mission requirements.

Hardware Design Description

The derivation and characteristics of the data handling subsystem hardware are shown in Table 5-5. The table briefly describes the source and type of hardware employed in addition to presenting the mass, power, and volume characteristics for both the probe bus and orbiter spacecraft.

Three of the data handling units are planned to be built by Hughes. Of these units, the remote multiplexer and dual PCM encoder units are existing OSO units, whereas the dual telemetry processor is a new design that employs OSO circuits to a large degree. The remote multiplexer and dual telemetry processor employ a large amount of custom LSI circuitry as a means to reduce mass.

The data storage unit is planned to be purchased from Electronic Memories. A previously designed core memory unit, of the company's SEMS series, will be modified to meet the storage and interface requirements.

TABLE 5-4. PROBE BUS/ORBITER SPACECRAFT DATA HANDLING SUBSYSTEM FUNCTIONAL CHARACTERISTICS SUMMARY

Parameter	Characteristics	
	Probe Bus	Orbiter
Word size	8 bits	
Number of data inputs	224	
Analog-to-digital conversion accuracy	±0.4% (8 bits)	
Data storage capacity	None	393 kilobits
Data bit rates	8 to 2048 bps	
Downlink data format		
Subframe length	32 words	
Telemetry Frame length	512 words, 16 subframes	
Number of preprogrammed (ROM) 32 word subframes	16 bus unique	16 orbiter unique
Number of downlink formats	All combinations of from 1 to 16 subframes in groups of 16 are possible; actual combinations will be selected by ground command	
Error code	Convolutional, length 32, rate 1/2	
Modulation type	PCM/PSK	
Subcarrier frequency, Hz	(TBD)	(TBD)

TABLE 5-5. MODULAR DATA HANDLING SUBSYSTEM – HARDWARE DERIVATION AND CHARACTERISTICS

Unit	Hardware Derivation: Thor/Delta Probe Bus	Hardware Derivation: Thor/Delta Orbiter	Hardware Characteristics*: Mass (Weight), kg (lb)	Hardware Characteristics*: Power, W	Hardware Characteristics*: Volume, cm^3 (in^3)
Remote multiplexer	OSO (seven single units; existing LSI; printed circuit boards)	Same	1.4 (3.0)	0.3	970 (59)
Dual PCM encoder	OSO (Dual unit; existing MSI; MICAM, printed circuit boards, and modules)	Same	2.8 (6.3)	2.6	3,500 (214)
Dual telemetry processor	New design (Uses 70 percent OSO circuits & new LSI; MICAM and printed circuit boards)	Same (modified ROM formats)	1.6 (3.5)	5.7	2,590 (158)
Data storage unit	Not required	Core - POV (modified electronic memories design)	--- --- 4.1 (9.0)	--- 0.3	--- --- 2,290 (140)
		Probe bus totals	5.8 (12.8)	8.6	7,060 (431)
		Orbiter totals	9.9 (21.8)	8.9	9,350 (571)

*Where two sets of values are given, the bottom line is for the orbiter spacecraft.

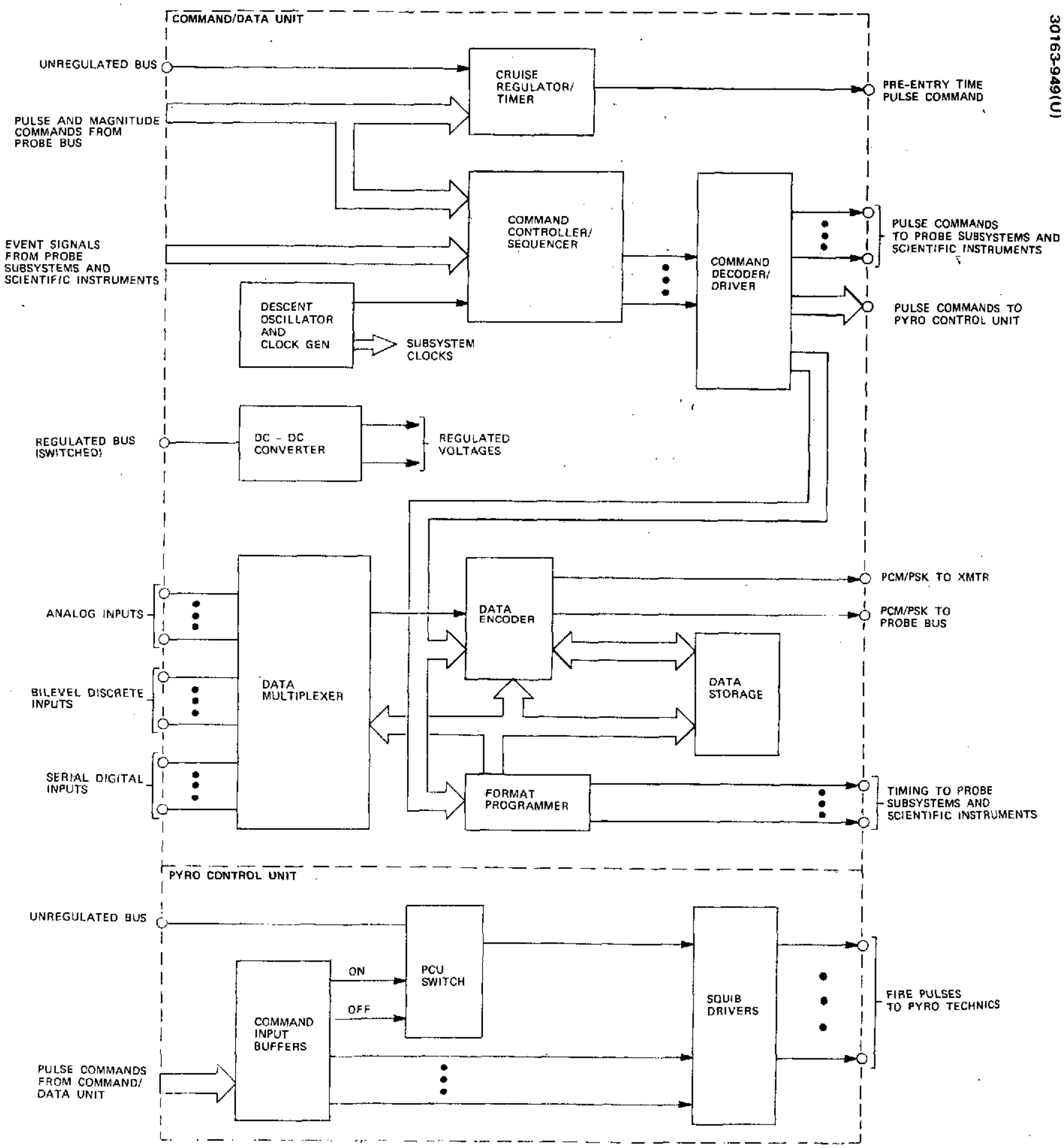


FIGURE 5-3. LARGE AND SMALL PROBES COMMAND AND DATA HANDLING SUBSYSTEM

5.2 PROBE COMMAND AND DATA HANDLING SUBSYSTEMS

Functional Description

The command functions and the data handling functions are combined into a single subsystem in both the large and small probes. The subsystem is divided into two control items or units: the command/data unit and the pyro control unit. Figure 5-3 shows a block diagram of the subsystem.

The command/data unit:

- Provides an accurate timer for initiation of the pre-entry sequence
- Initiates pulse commands from real-time events and stored command sequences
- Multiplexes and encodes analog and digital input data to form a serial PCM/PSK bit stream
- Provides multiple stored formats and data rates for downlink data transmission
- Provides for data storage during entry blackout

The pyro control unit provides high current switching capability for firing of pyrotechnic devices.

Table 5-6 summarizes the functional characteristics of the command and data handling subsystems of the large and small probes.

Hardware Design Description

A summary of the hardware derivation and the mass, power and volume characteristics of the large and small probe subsystems is presented in Table 5-7. New design was planned for both probes since appropriate existing hardware was not available which would meet the performance requirements plus the environmental and packaging constraints.

From an electrical design standpoint, minimization of mass and volume was accomplished by designing for minimum parts count, primarily by use of LSI. From a mechanical design standpoint, packaging techniques were investigated to ensure the ability to withstand high deceleration forces during entry into the Venus atmosphere; also, conformal packaging was considered to make optimum use of the spherical envelope of the probe pressure vessel.

There is much commonality between the large and small probe designs due to their functional similarity; many circuits are identical for

TABLE 5-6. LARGE/SMALL PROBE COMMAND AND DATA HANDLING SUBSYSTEMS FUNCTIONAL CHARACTERISTICS SUMMARY

Parameter	Characteristic: Large Probe	Characteristic: Small Probe
Command		
Cruise timer		
Range	20 days	
Stability (0° to $40^{\circ}C$)	1×10^{-5} (17.3 sec in 20 days)	
Resolution	1.6 sec	
Descent timer		
Range	13.6 minutes (between recycle)	
Stability (-20° to $+65^{\circ}C$)	5×10^{-5}	
Resolution	0.1 sec	
Command mode	Stored	
Command type	Pulse	
Command initiation	Time or Event	
Number of event inputs	8	
Number of commands	48	32
Total command executions	128	64
Number of pyrotechnic drivers	4 redundant pairs, multiple 2 nonredundant, multiple 10 nonredundant, single	1 redundant pair, multiple 1 nonredundant, multiple
Maximum delay between concurrent fire pulses	0.1 ms	
Number of pyro initiators fired (capability)	40	9
Data handling		
Word size	10 bits	
Number of analog inputs		
10 bit accuracy	32	8
8 bit accuracy/10 bit resolution	44	20
Number of digital inputs		
10 bit serial	8	4
Bilevel discrete	28	20
Data storage capacity	4096 bits	512 bits
Number of data formats	3	2
Data bit rates	276 or 184 bps	16 bps
Error coding	Convolutional, length 32, rate 1/2	
Modulation type	PCM/PSK	
Subcarrier frequency	4416 Hz	256 Hz

TABLE 5-7. LARGE AND SMALL PROBES COMMAND AND DATA HANDLING SUBSYSTEM HARDWARE DERIVATION AND CHARACTERISTICS

Unit	Thor/Delta Hardware Derivation	Hardware Characteristics			
		Mass (Weight), kg	(lb)	Power Cruise/Descent, W	Volume, cm^3 (in^3)
Large probe command/data unit	New design (Printed circuit boards and modules; LSI)	1.13	(2.5)	0.06/4.2	1,426 (87)
Large probe pyro control unit	New design (Uses 50 percent of probe bus circuits; modules)	1.00	(2.2)	(Transient only)	1,196 (73)
Total		2.13	(4.7)	0.06/4.2	2,622 (160)
Small probe command/data unit	New design (Uses 50 percent of large probe circuits; printed circuit boards and modules; LSI)	1.00	(2.2)	0.06/4.2	1,180 (72)
Unit probe pyro control	New design (Uses 80 percent large probe circuits; modules)	0.22	(0.5)	(Transient only)	164 (10)
Total		1.22	(2.7)	0.06/4.2	1,344 (82)

both. The experiment interface specification is identical for both probes and, in fact, is nearly identical to that of the probe bus/orbiter spacecraft; a summary of the interface specification can be found in Section 4 of this volume. Functional blocks in Figure 5-3 that are identical for both the large and small probes include the cruise regulator/timer, data encoder, and the regulator and dc-dc converter. In addition, circuit elements such as multiplexer switches, memory storage devices, command decoders and drivers, timing circuits and squib drivers are identical but are used in smaller numbers in the small probe. This results from the difference in magnitude of requirements between the two probes; e.g., number of data inputs, storage capacity, number and type of data formats, number of stored command sequences and the number of pulse command outputs. The pyro control unit in the small probe is considerably smaller than that of the large probe, although the individual switching circuits are the same for both.

Large Scale Integration

Custom developed LSI circuits are used to minimize mass and volume in the probe design. They are used only in the command/data unit, since they do not apply as well to the circuits in the pyro control unit. A total of three chips would be developed for use in both probes. Each LSI chip is used once in each probe. Two of the three types use PMOS technology; the third, which is used for the cruise timer, uses CMOS technology for ultra-low power dissipation. This use of LSI has reduced the overall command and data handling subsystem mass and volume by approximately 40 percent, as compared to the use of IC and MSI technology.

6. ATLAS/CENTAUR BASELINE DESCRIPTION

This section describes the Atlas/Centaur baseline designs for the command and data handling subsystems on the probe bus, the orbiter, the large probe, and the small probe vehicles. Table 6-1 presents a summary of mass, power, and volume characteristics for the subsystems on all four spacecraft.

This baseline differs in a number of respects from the Thor/Delta baseline described in Section 5. A major difference is that custom LSI circuit technology is employed to a much lesser degree in the Atlas/Centaur version of these subsystems since they are not as mass and volume constrained as those for the Thor/Delta configuration; this results in lower cost equipment. Also, several functional differences are due to program level changes made subsequent to the decision to use the Atlas/Centaur launch vehicle. The program changes affecting these subsystems were primarily:

1) The decision to utilize the same launch opportunity for both the multiprobe and orbiter missions, resulting in two spacecraft in transit at the same time

2) The decision to use a type II trajectory for the orbiter in order to allow separation of the launch windows

3) Revised science payload requirements

The subsystem baselines described in this section reflect these changes and the corresponding new requirements.

Subsystem requirements affected by the above program decisions are described in subsection 6.1.

Subsection 6.2 discusses the design rationale and tradeoffs for the Atlas/Centaur baseline and reflects the impact of these new requirements. Since the tradeoff studies discussed in Section 4 of this volume were primarily Thor/Delta oriented, each is reviewed to show its correlation and the applicability of its results to the Atlas/Centaur design; changes in the results and conclusions of those studies caused by the new requirements are noted.

TABLE 6-1. ATLAS/CENTAUR COMMAND AND DATA HANDLING SUBSYSTEM CHARACTERISTICS SUMMARY

Subsystems	Characteristics				
	Mass		Power, W	Volume	
	kg	(lb)		cm^3	(in^3)
Probe bus					
Command subsystem	9.8	(21.7)	6.9	15,060	(919)
Data handling subsystem	7.8	(17.2)	8.7	11,180	(682)
Totals	17.6	(38.9)	15.6	26,240	(1,601)
Orbiter					
Command subsystem	9.8	(21.7)	6.9	15,060	(919)
Data handling subsystem	16.9	(37.2)	11.7	20,810	(1,270)
Totals	26.7	(58.9)	18.6	35,870	(2,189)
Large probe command and data handling subsystem	3.5	(7.8)	Cruise/descent 0.04/8.3	4,140	(253)
Small probe command and data handling subsystem	2.7	(5.9)	0.04/5.8	2,930	(179)

The baseline designs for the probe bus and orbiter spacecraft command and data handling subsystems are described in subsection 6.3. These designs are practically identical and make extensive use of previously developed OSO hardware. As with the Thor/Delta design, the major difference is the addition of a data storage unit on the orbiter spacecraft; also, minor differences exist in the telemetry data formats.

Subsection 6.4 describes the command and data handling subsystem baselines for the large and small probes; these designs are common to a high degree. Essentially, the small probe subsystem is a subset of the large probe; although the designs are very similar, the small probe requires fewer multiplexer elements for gathering data, fewer command decoding and command outputs, fewer memory elements for data storage, and fewer drivers for firing of pyrotechnic initiators.

6.1 SUBSYSTEM REQUIREMENTS IMPACT

Requirements for command and data handling subsystem performance have been modified due to the recent changes in the baseline mission which were discussed in the preceding paragraphs. The changes are summarized as follows:

1) Uplink bit rate — bus changed from selectable 1, 2, 4, or 8 bps to a single 4 bps. This is possible due to changes in the link performance which eliminate the need for bit rates less than 4 bps. The 8 bps rate was not required and was eliminated to minimize changes to existing hardware.

2) Command memory size — has changed from 64 to 85 24-bit words. This change was required due to increased commanding necessary at periapsis for the radio occultation experiment antenna positioning.

3) Receiver reverse unit — The RRU algorithm has been changed to accommodate switching from the high gain antenna to the omni antennas in the event of an uplink failure.

4) Pyrotechnic damages — have been necessary to accommodate changes in the science experiments and mechanisms.

5) Orbiter data storage size — has increased from 393 kilobits to 1 megabit due to increased science instrument data rates at periapsis.

6) Simultaneous real time and stored data processing requirements — has been necessary due to the volume of data collected at orbiter periapsis. Downlink capability is limited to 128 bps through the last half of the orbiter mission, which requires storing of the remaining science periapses requirement at approximately 512 bps.

7) Additional data formats and bit rates — were required to fit the orbiter periapsis science data storage requirements to realizable data storage devices. Bit rates have been added to the small probe to accommodate additional science data.

8) Command and data channel allocations — have changed for all subsystems due to design modifications and the minimum changes.

9) Probe cruise timer — timing requirement has changed from 20 to 23 days. This change was implemented subsequent to the mid-term design review due to the desire to spread out the probe release sequence of events and provide longer periods of uninterrupted tracking to reduce entry dispersions.

10) Pressure/acceleration switches — a bookkeeping change was made to include these switches in the probe command and data subsystem from the mechanisms subsystem.

11) Subcarrier frequencies — have changed for the large and small probe to accommodate the transponder interface requirements.

12) Data storage — for the large probe has decreased slightly due to elimination of the shock layer radiometer.

An overall summary of command assignments, data channel assignments, and command and data system requirements has been tabulated. Multiprobe bus command requirements are given in Table 6-2; assignments are shown in Table 6-3. Orbiter spacecraft command requirements are given in Table 6-4; assignments are shown in Table 6-5. Large and small probe command requirements are presented in Table 6-6.

TABLE 6-2. MULTIPROBE BUS COMMAND REQUIREMENTS

Demodulate, decode, distribute ground commands
Store commands for later execution, 17 words minimum storage
PCM/PSK/PM or PCM/FSK/PM modulation
4 bps command rate
36-bit command word
Pulse commands
144 engineering
14 instrument
Magnitude commands
8 engineering
Interlocked commands
5 engineering
2 science instrument
Threshold: 1×10^{-5} BER
Probability of executing a false command at threshold: 1×10^{-9}

TABLE 6-3. MULTIPROBE BUS COMMAND ASSIGNMENT SUMMARY

Subsystem	Pulse	Magnitude	Interlocked
Command	6	1	0
Communications	21	0	0
Control/propulsion	17	1	1
Data handling	6	2	0
Power	78	0	0
Probes	16	4	4
Science	14	0	2
Thermal	0	0	0
Total	158	8	7

TABLE 6-4. ORBITER SPACECRAFT COMMAND REQUIREMENTS

Demodulate, decode, distribute ground commands

Store commands for later execution, 85 words minimum storage

PCM/PSK/PM or PCM/FSK/PM modulation

4 bps command rate

36-bit command word

Pulse commands

124 engineering
43 instrument

Magnitude commands

5 engineering
2 instrument

Interlocked commands

2 engineering
2 instrument

Threshold: 1×10^{-5}

Probability of executing a false command at threshold: 1×10^{-9}

TABLE 6-5. ORBITER COMMAND ASSIGNMENT SUMMARY

Subsystem	Pulse	Magnitude	Interlocked
Command	6	1	0
Communications	19	0	0
Control/propulsion	27	2	2
Data handling	9	2	0
Power	63	0	0
Science	43	2	2
Thermal	0	0	0
Total	167	7	4

TABLE 6-6. SUMMARY OF PROBE COMMAND REQUIREMENTS

Subsystem	Large Probe	Small Probes
Data handling	11	11
Mechanisms	16	3
Power	8	8
Science	6	4
Total	41	26

Probe bus handling requirements are shown in Table 6-7; telemetry channel requirements are presented in Table 6-8. Orbiter spacecraft handling requirements are given in Table 6-9; telemetry channel requirements are shown in Table 6-10. A summary of probe telemetry channel requirements is given in Table 6-11.

TABLE 6-7. PROBE BUS DATA HANDLING REQUIREMENTS

Multiplex, format, encode, and modulate spacecraft and experiment data
PCM/PSK/PM modulation
Convolutional encoding: K = 32; R = 1/2
8 to 2048 bps data rates
Minor frames: 16
Major frames: 16
Engineering data channels 107 analog 39 serial digital 0 discrete
Science instrument data channels 7 analog 6 serial digital 0 discrete

TABLE 6-8. PROBE BUS TELEMETRY CHANNEL REQUIREMENT SUMMARY

Subsystem	Analog	Digital	Total
Command	0	6	6
Communications	12	2	14
Control/propulsion	11	11	22
Data handling	4	9	13
Power	40	6	46
Probes	22	5	27
Science	7	6	13
Thermal	18	0	18
Subtotal	114	45	159
Spare			33
Total			192

TABLE 6-9. ORBITER SPACECRAFT DATA HANDLING REQUIREMENTS

Multiplex, format, encode, and modulate spacecraft and experiment data
Store data for later transmission to earth
PCM/PSK/PM modulation
Convolutional encoding: K = 32; R = 1/2
8 to 2048 bps data rates
Minor frames: 16
Major frames: 16
Engineering data channels
100 analog
34 serial digital
0 discrete
Science instrument data channels
26 analog
17 serial digital
0 discrete
Storage capacity: 1 megabit

TABLE 6-10. ORBITER SPACECRAFT TELEMETRY CHANNEL REQUIREMENT SUMMARY

Subsystem	Analog	Digital	Total
Command	0	6	6
Communications	12	2	14
Control/propulsion	24	11	35
Data handling	4	9	13
Power	42	6	48
Science	26	17	43
Thermal	18	0	18
Subtotal	126	51	177
Spare			15
Total			192

6.2 DESIGN RATIONALE AND TRADEOFFS

This subsection describes the rationale used in the design of the command and data handling subsystems for the Atlas/Centaur launch vehicle in each of the probe bus/orbiter spacecraft and the large and small probes. In particular, any tradeoff studies which were performed subsequent to, or resulting from, recent changes to the Pioneer Venus program requirements will be discussed. In some instances, the new requirements may have affected the criteria used in performing the studies reported in subsections 4.1 and 4.2, thereby affecting the results or conclusions of those studies. Each of the studies reported previously will be discussed, with any changes in the results noted.

The principal effect on the command and data handling subsystem resulting from the decision to use the Atlas/Centaur launch vehicle is that the development of custom large scale integrated circuits is no longer justified from a cost standpoint since mass and volume are no longer as severely limited as for the Thor/Delta vehicle. Thus, mass and volume are higher than for the Thor/Delta design in both the spacecraft and the probes.

TABLE 6-11. SUMMARY OF PROBE TELEMETRY CHANNEL REQUIREMENTS

Subsystem	Analog	Serial Digital	Binary Discrete
Large probe			
Data handling	4	3	2
Instrumentation	7	0	0
Power	5	2	0
Science	16	6	0
Thermal	5	0	0
Total	37	11	2
Small probe			
Data handling	4	3	2
Power	4	1	0
Science	7	5	0
Thermal	4	0	0
Total	19	9	2

The revised science requirements and new mission set caused many subsystem requirements to change, necessitating a redefinition of data formats, data storage requirements, downlink bit rates, and so forth. Of particular significance was a considerable increase in the data storage requirements for the orbiter spacecraft.

Command Subsystem Design Changes and Rationale

Changes to the command subsystems in the probe bus/orbiter spacecraft and the large and small probes are summarized below.

For all four vehicles, the planned use of custom LSI was abandoned to achieve a lower cost design in light of the less severe mass and volume constraints upon the Atlas/Centaur design.

For the probe bus/orbiter spacecraft:

1) Subsystem units which were previously dual units were separated into two identical redundant units in order to make them identical to existing OSO hardware; this results in a lower cost design. The slight increase in mass and volume is felt to be acceptable in view of the less stringent mass constraints upon the Atlas/ Centaur design.

2) The real time uplink bit rate was changed from a selectable 1, 2, 4, or 8 bits bps to a single 4 bps in order to minimize modifications to existing command demodulator hardware; both demodulator candidates presently under consideration operate at the higher rate. The change was permitted as a result of improvements to the uplink performance.

3) The circuitry used to control the antenna/receiver select switch was relocated from the demodulators to the central decoder since its location in the demodulator required a change to existing hardware. In addition, a new algorithm for selecting the receiver/antenna connection would have required additional interfacing between the central decoder and the demodulator; relocating the control circuitry now results in a minimum interface.

4) The command memory was changed from a single input (entry point) shift register to a multiple entry point shift register in order to facilitate reloading of incorrectly received commands. The change in command storage requirements from 64 to 85 words did not impact the command memory hardware per se, since the 85 word capability previously existed; however, the probability of correctly loading 85 words is not as high as with 64 words, thus making more desirable the change to the multiple entry point memory. This change, in turn, necessitated adding a 16-bit shift register to the existing 2048-bit command memory in order to have an integral number of commands (86) in memory.

5) The requirement for pyrotechnic drivers was revised from 6 to 5 on the multiprobe bus and from 3 to 5 on the orbiter as a result of new science and engineering requirements for the two missions. This enables a common pyro control unit design for both spacecraft, resulting in a considerable reduction in costs associated with product design and documentation for a separate control item.

6) The number of spare pulse commands has increased and the number of spare magnitude commands has decreased as a result of revised command requirements.

For the probes (command portion of the command and data handling subsystem):

1) A bus voltage regulator was added to the command and data handling subsystem as a result of the decision to change to an unregulated bus on the probes.

2) The range of the probes' cruise timer was increased to 24 days to allow for a longer period from probe separation to entry.

3) The use of a switching regulator for the cruise timer was reevaluated due to the change to a higher battery voltage in the probes.

4) The number of spare pulse command outputs was decreased due to an increase in command requirements.

5) The number of pyrotechnic drivers was increased to accommodate new science payload requirements.

6) The pyro control unit (PCU) ON/OFF switch was relocated from the power subsystem to the PCU; this relay switch now performs the dual function of arming the PCU switches and blocking leakage current during the probe cruise period.

7) Pressure and acceleration switches were reassigned to the command and data handling subsystem and are now reflected in subsystem hardware characteristics.

Command Subsystem Trade Studies

Each of the trade studies reported in subsection 4.1, as it may be affected by the changes outlined above, is discussed below.

Command Interface Methods

The interface definition summarized in subsection 4.1 was not affected by the changes in program requirements.

Prevention of Spacecraft Inadvertent-Irreversible Command Execution

The results of the original study were not affected by the changes in program requirements.

Spacecraft Command Storage Analysis

Recent changes caused the command storage requirement to change from 64 to 85 command words. An additional requirement was placed on the command memory such that the probability of loading the memory correctly would be greater than 0.996 in a 30 min period exclusive of verification and round trip delay time. Because of the time limitation on loading and correcting the command memory, it was decided that a random access memory (RAM) or a shift register with multiple entry points would be required. Investigation of both techniques showed that the long shift register with multiple entry points would require less hardware and power than a RAM. The study concluded that the long MOS shift register memory is still applicable. Additional circuitry, however, would be needed to provide more than one entry point.

Probe Cruise Timer

The decision to use the Atlas/Centaur launch vehicle reinforces the selection of the switching regulator/1 MHz oscillator made in the original study since the mass constraint is not as severe as it was previously. The selected approach consumes 30 mW, whereas the low frequency alternate consumes 15 mW. The mass penalty paid for the higher power circuit is approximately 0.12 kg (0.27 lb) per probe battery or an overall mass difference of 0.48 kg (1.1 lb) for the total battery complement. This is felt to be an acceptable tradeoff in light of the advantages of better stability and lower technical risk.

A 30 mW savings was realized by using a switching instead of a series regulator with the 1 MHz oscillator, resulting in an overall mass reduction for the total probe battery complement of almost 0.9 kg (2 lb). If the probe battery voltage were increased to 28 V, as might be the case if an unregulated bus concept were adopted, the series regulator approach would consume approximately 160 mW. Then the battery mass savings realized by using the switching regulator approach at 40 mW dissipation (up from 30 mW due to lower efficiency at 28 V) would be an even more significant 3.9 kg (8.5 lb). Thus, it is felt that the use of a switching regulator and the 1 MHz crystal oscillator, as proposed in the original study, still represents the optimum design approach.

Receiver Reverse Unit

The discussion on the receiver reverse unit functions included in subsection 4.1 was based upon the current Atlas/Centaur design. It therefore remains unchanged.

Data Handling Subsystem Design Changes and Rationale

Changes to the data handling subsystems in the probe bus/orbiter spacecraft and the large and small probes are summarized below.

For all four vehicles, the planned use of custom LSI was abandoned to achieve a lower cost design in light of the less severe mass and volume constraints upon the Atlas/Centaur design.

For the probe bus/orbiter spacecraft:

1) Subsystem units which were previously dual units were separated into two identical redundant units in order to make them identical to existing OSO hardware; this results in a lower cost design. The slight increase in mass and volume is felt to be acceptable in view of the less stringent mass constraints upon the Atlas/ Centaur design.

2) The remote multiplexers, seven of which were previously required, are reduced to three dual units in order to be identical to existing OSO hardware; the reduction in the number of data inputs from 224 to 192 still allows adequate spare inputs, and a lower cost, lower mass design is achieved.

3) The orbiter data storage capacity was increased from 393 kilobits to 1 megabit because of increased storage requirements. In addition, the data storage unit is now divided into two identical units (each 524 kilobits) in order to have graceful degradation of data storage capability; the added reliability justifies the mass penalty for the pair of storage units (about 70 percent greater mass than a single unit of the same capacity) in light of the increased mass allowances of the Atlas/Centaur design.

4) A requirement for the orbiter mission of simultaneous dual path processing of data into two different formats at different bit rates resulted in about a 30 percent increase in the complexity of the telemetry processor. Dual path processing is required in order to be able to simultaneously transmit real time data at the maximum downlink capability at periapsis (128 bps) and to store the remainder of the required science data for transmission when the link is not so constrained.

5) The scheme for programming and selection of formats was changed. Initially, the capability for ground programming of two of several possible formats and selection between the two formats by real time or stored command was provided. Presently, 16 different formats are preprogrammed into read-only memory (ROM) elements, and selection of the desired format can be made by real time or stored command. This allows rapid changing among several formats, which is required during certain phases of the orbiter mission. Some flexibility of format programming is lost; however, the 16 format capability exceeds the maximum requirement of 12 so that sufficient flexibility is still felt to exist. This change permitted about a 10 percent reduction in the complexity of the telemetry processor.

6) An additional bit rate of 682.7 bps (2048/3) was added to the data handling subsystem capability in order to minimize the orbiter data storage requirement at periapsis. Previously, all bit rates in binary ratios from 8 to 2048 bps were provided. Provision for the new bit rate requires a small amount of additional circuitry to generate the signal, to command the change to that rate, and to signal users of that bit rate mode.

For the probes (data handling portion of the command and data handling subsystem):

1) The number of formats and structure of formats required for downlink data transmission and data storage were changed for both large and small probes.

2) The data storage capacity was reduced in the large probe because of reduced storage requirements; in addition, the data storage element was changed from a RAM to a static shift register.

3) The number of bit rates required for downlink data transmission and data storage has been increased.

4) The number of data inputs required has been revised, allowing a reduction in multiplexer hardware.

5) The biphase modulating subcarrier frequencies have been changed for both probes to accommodate the transponder interface.

Data Handling Subsystem Trade Studies

Each of the trade studies reported in subsection 4.2, as it may be affected by the changes outlined above, is discussed below.

Telemetry Data Recovery Analysis

The material found in subsection 4.2 was written with the current Atlas/Centaur parameters included.

Data Handling Interface Methods

The interface definition summarized in subsection 4.2 was not affected by the recent changes in program requirements.

Orbiter Data Storage Analysis

The data storage requirements for the orbiter spacecraft increased from approximately 393 kilobits to about 1 megabit as a result of the new science requirements and the new mission set. The original study was based upon memory capacities from 100 kilobits to 2 megabits; thus, the results of that study still apply to the present memory requirement.

Since magnetic tape recorders were not considered in the original study, it seemed appropriate to make a cursory investigation of their possible application in light of the increased storage requirements. Data was gathered on more than 50 tape recorders and their characteristics were reviewed. It was determined that magnetic core is still preferred over a magnetic tape recorder since a core memory system has reliability, mass, and cost advantages for this application. This is primarily due to the fact that a tape recorder would need an additional buffer memory in order to accommodate the various bit rates required during data gathering and playback; a core memory system can handle these multiple bit rates in a straightforward manner, has no moving parts, and does not require additional interface buffering equipment.

Probe Data Storage Hardware

The data storage requirement on the large probe was reduced from 4096 to 2048 bits as a result of a change in the probe descent profile and of the new science requirements. The conclusions made in the original study are still applicable. However, the 256-bit memory device chosen in that study will not be used because a preferred device will be space qualified for use on the probe bus/orbiter spacecraft; it can also be used on the probes and will allow a reduction in mass, power, and volume. This device is a low power, 512 bit static shift register. Although four of the devices will be needed to make up the large probe memory, the individual package is quite small, and the serial-in/serial-out data transfer is ideally suited for storing the serial bit stream in the large and small probes.

Probe Stored Data Playback Techniques

The results of the study to determine the optimum technique for playing back stored data in the probes were not affected by the recent changes in program requirements.

Probe Multiple Data Formats

Since the original study to determine the impact upon the probe data handling subsystems of providing two or more data formats, the need for multiple formats has been well established from an overall system standpoint. Input data sampling requirements are now well enough defined that it is possible to more thoroughly assess the impact of providing multiple formats. These new assessments are described below and can be used as guidelines for any additional format changes that may develop in the future.

The Harris HPROM 1024 read only memory (ROM) chosen in the original study, a 1024 bit bipolar TTL-compatible device, is still felt to be the optimum choice for the format programming application. The ROM is organized as 256 words X 4 bits. In the current large probe baseline, provision is made for 96 data inputs. However, less than 12 of these inputs account for over 90 percent of the content of each data format. The remainder of the inputs are "subcommutated" or sampled at a very low rate. Subcommutation results in complication of the data formatting design; however, with the use of ROMs, complication of hardware is minimized.

The basic telemetry frame requires two ROMs in order to provide an 8 bit output word. Five of the 8 bits are used to address up to 32 data inputs. The other three bits are used to tag a ROM output address word. The tag identifies whether the word defines an analog, serial digital, or bilevel discrete data input, whether a subcommutated input will be inserted into the basic frame, or whether a sync word will be inserted.

Two additional ROMs are required for subcommutated inputs. Six of the eight output bits are used to address up to 64 subcommutated inputs; the remaining two are tag bits.

Two 256 word X 4 bit ROMs can either accommodate a basic format of 256 words or other possible combinations such as two formats of 128 words, four formats of 64 words, etc. Likewise, the ROMs used for subcommutated inputs can accommodate similar subcom frame formats.

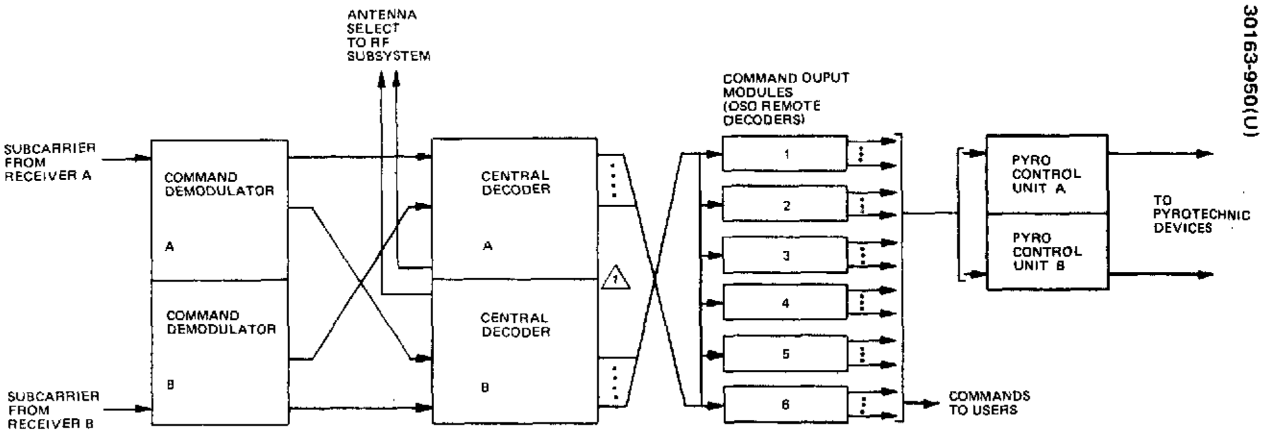
Thus, it can be seen that with four ROMs, which is the minimum number possible if subcommutation is necessary, four relatively complex data formats could be provided. Each of these formats could have a 64 word basic frame and a 64 word subcom frame; the maximum possible total data frame for each format would be 4096 words (64 basic frames, each containing one word of the subcom frame).

Providing up to four data formats can be done with minimal impact upon the probe design if the basic frame length and subcom frame length are kept to 64 words each. But, for example, if two basic frame formats of 128 words each were required, in general ROM elements would have to be added to provide more than those two formats.

Although larger ROM elements are available, including a 512 word X 8 bit MOS device which is presently space qualified by Hughes, the Harris HPROM 1024 is preferred. In addition to being space qualified, the Harris device is field programmable so that format changes, if necessary, can be made much later in the program than with the larger device, which is programmed early in its fabrication process. Moreover, the HPROM 1024 is TTL compatible and packaged in a relatively small 16 pin unit, so overall packaging efficiency and cost are not significantly different than with the larger device.

Probe Multiple Data Rates

The results of this study, which was conducted to determine the impact upon the probe data handling subsystem of providing more than one data rate for downlink data transmission, were not affected by the recent changes in program requirements.



1 INTERFACE IS FULLY CROSS - STRAPPED; EACH CENTRAL DECODER COMMUNICATES WITH EACH COMMAND OUTPUT MODULE

FIGURE 6-1. MODULAR COMMAND SUBSYSTEM

30163-950(U)

6.3 SPACECRAFT COMMAND AND DATA HANDLING SUBSYSTEMS

The Atlas/Centaur command and data handling subsystems are functionally quite similar to those described for Thor/Delta in Section 5. However, there are a number of significant hardware design differences. These functional and hardware differences have been previously described in section 6 and subsection 6.2.

As with the Thor/Delta, the Atlas/Centaur command and data handling subsystems are highly modular in nature and are essentially identical on the probe bus and orbiter spacecraft. The few differences that exist are explained in the following discussions.

Probe Bus/Orbiter Command Subsystem Functional Summary

Figure 6-1 shows a block diagram of the command subsystem. It consists of four basic types of units, each employing a complete dual redundancy: command demodulators, central decoders, six single remote decoders, and pyro control units. The following is a brief description of the major functions performed by each of these units.

Command Demodulator

1) Receives and demodulates subcarrier into command data and clock signals

2) Relays the data and clock signals to the central decoder

Central Decoder

1) Decodes and processes the incoming command data

2) Performs a polynomial code check for command verification

3) Provides storage for time dependent command sequences

4) Initiates execution of the commands at the proper time

5) Distributes commands to the proper remote decoder

6) Provides receiver reverse switching to select the connection combination between the receivers and antennas

Remote Decoder

1) Recognizes and decodes the appropriate command addresses

2) Receives and decodes command data into pulse or magnitude commands

3) Distributes pulse commands and magnitude data to spacecraft users (and to probes on the probe bus)

Pyro Control Unit

1) Receives interlocked pulse commands from remote decoders

2) Provides high current switching to fire pyrotechnic devices

Table 6-12 summarizes the basic functional characteristics of the command subsystem for both the probe bus and orbiter spacecraft. As indicated in the table, the characteristics are identical.

TABLE 6-12. PROBE BUS/ORBITER SPACECRAFT COMMAND SUBSYSTEM FUNCTIONAL CHARACTERISTICS SUMMARY

Parameter	Probe Bus and Orbiter Characteristics
Command mode	Real time or stored
Command type	Pulse or magnitude
Command initiation	Ground command, timer or event
Command uplink	
Command word size	36 bits (includes 7 bit error code)
Bit rate	4 bps
Modulation	PSK
Probability of false command	1×10^{-9}
Number of pulse command outputs	192
Number of magnitude command outputs	12
Word size of magnitude commands	16 bits
Command storage capacity	2064 bits (86 words maximum)
Resolution of command executions	±125 ms (stored mode)
Number of pyrotechnic drivers	5 redundant pairs
Maximum delay between concurrent fire pulses	0.1 ms
Number of pyrotechnic initiators fired (capability)	30

Probe Bus/Orbiter Command Subsystem Hardware Design Summary

A summary of the hardware derivation and characteristics is shown in Table 6-13. In addition to presenting the mass, power, and volume characteristics for both the probe bus and orbiter spacecraft, the table briefly describes the source and type of hardware employed.

As indicated in the table, three of the units are planned to be built by Hughes and utilize OSO designs. Of these units, the central decoder and remote decoder employ custom LSI circuit technology as a means for reducing mass; the pyro control unit does not employ LSI since this technology is not as amenable to the high current pyrotechnic circuit applications.

The command demodulator is planned to be purchased from Motorola. This unit is a modification of the command demodulator that was developed for the Viking Orbiter program; it utilizes MSI circuit technology.

Probe Bus/Orbiter Command Subsystem Description

Command Demodulator

The command demodulator is a PSK demodulator and will be a purchased item. Initial indications are that the hardware will be a minor modification of one of the command demodulators used on the Viking program. It appears that, with such minor modifications, both the Viking lander and Viking orbiter command demodulators will meet the Pioneer Venus requirements. A simplified block diagram of the command demodulator is shown in Figure 6-2. The receiver subcarrier input frequency will be between 200 and 700 Hz. The data modulation will be at 4 bps. These frequencies are compatible with both the DSN and the Viking hardware.

The command demodulator is operated as follows. When the receiver in lock signal is present, the unit will be turned on. When the signal to noise ratio from the receiver is high enough to provide a bit error rate less than 10^{-5}, the command demodulator will lock on to both the subcarrier and data signals, regenerate the clock and data, and provide an additional signal to enable the central decoder. Telemetry outputs will be supplied to provide an indication of the signal to noise ratio and the uplink bit rate.

Central Decoder

The central decoder will be a new design using many of the circuits from the OSO-I central decoder. The central decoder performs three basic functions: 1) real time processing of uplink commands, 2) stored time processing of commands from the command memory, and 3) receiver reverse algorithm implementation. A block diagram of the central decoder is shown in Figure 6-3. Operation is as follows.

TABLE 6-13. ATLAS/CENTAUR MODULAR COMMAND SUBSYSTEM HARDWARE DERIVATION AND CHARACTERISTICS

Unit (Quantity)	Probe Bus and Orbiter Derivation	Characteristics*				
		Mass		Power, W	Volume	
		kg	(lb)		cm^3	(in^3)
Command demodulator (2)	Motorola - POV (Uses 98 percent Viking orbiter circuits; MSI; PC Boards)	2.5	(5.6)	3.0	3,080	(188)
Central decoder (2)	New design (Uses 50 percent OSO circuits; no new LSI; MICAM;* modules)	4.3	(9.4)	3.6	6,000	(366)
Command output module (6) (remote decoder)	OSO (Six single units; existing LSI)	1.8	(4.2)	0.3	1,720	(105)
Pyro control unit (2)	New design (Uses 70 percent existing circuits; modules)	1.2	(2.5)	0	4,260	(260)
Probe bus totals		9.8	(21.7)	6.9	15,060	(919)
Orbiter totals		9.8	(21.7)	6.9	15,060	(919)

*MICAM is the acronym for Hughes "microelectronic assembly method."

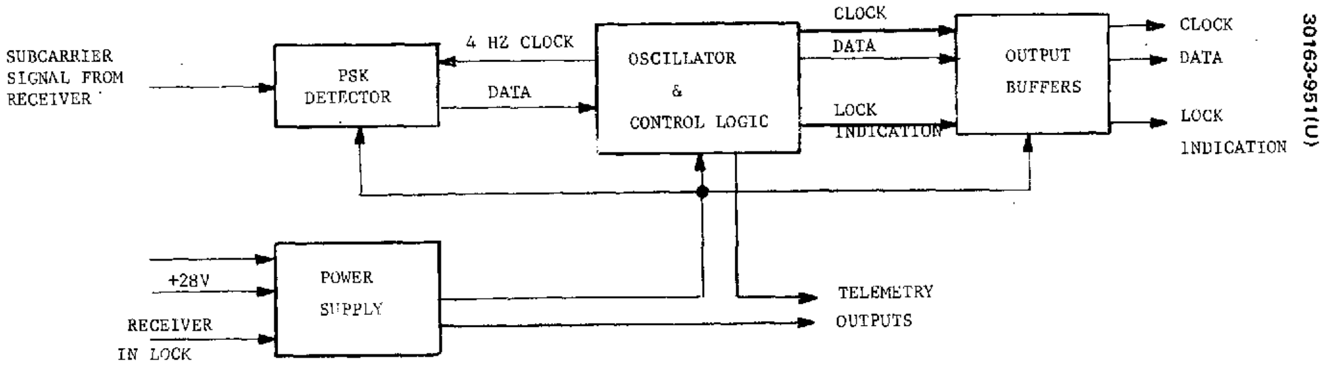


FIGURE 6-2. DEMODULATOR BLOCK DIAGRAM

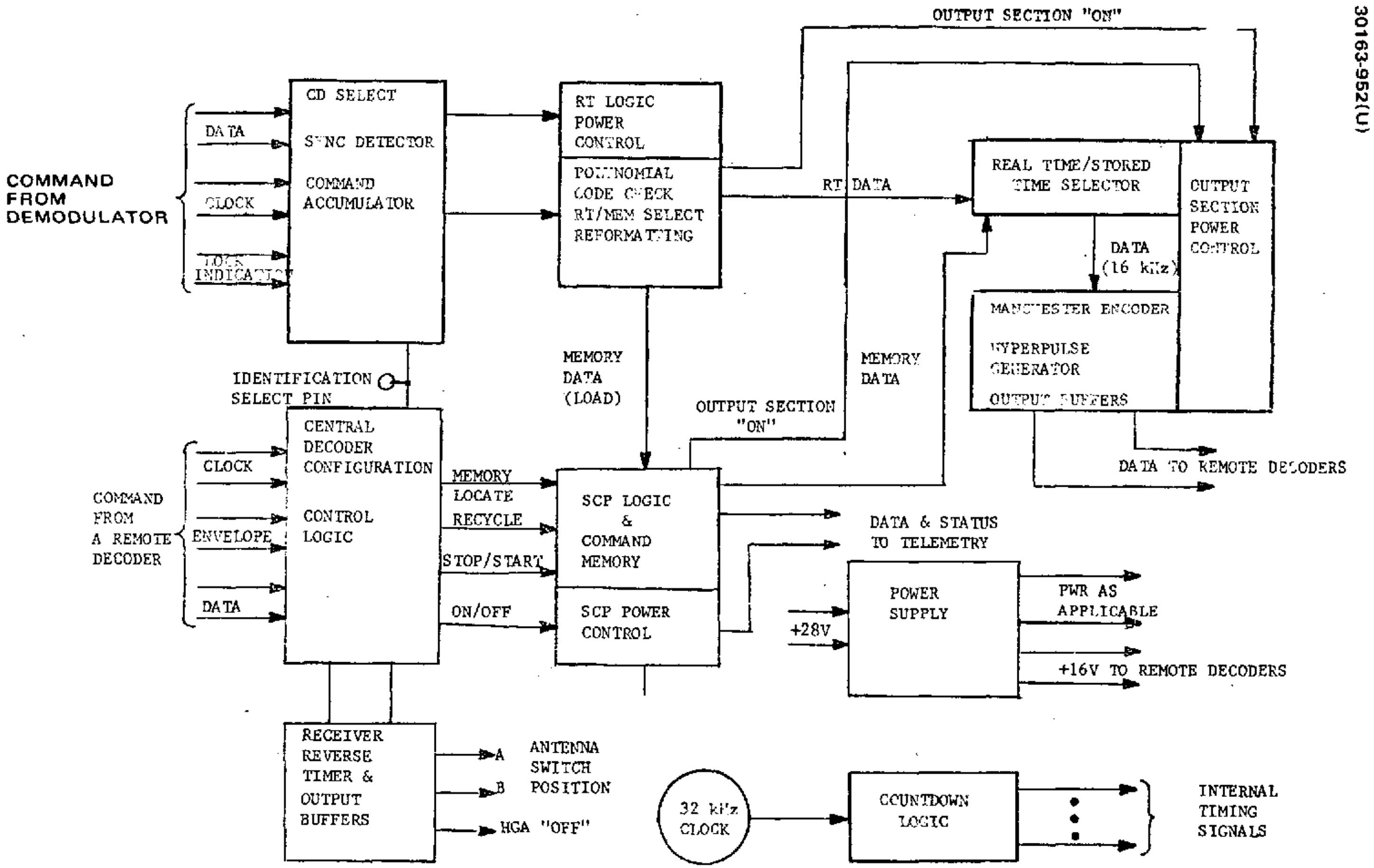


FIGURE 6-3. CENTRAL DECODER BLOCK DIAGRAM

Real Time Processing. When the in-lock signal from the command demodulator is active, the real time processing section will be turned on. The demodulated data is accumulated in a register at the uplink bit rate until the first sync pattern is detected. If the sync pattern does not appear within a specified number of bit times after turn on, the unit is turned off. Upon receipt of the sync pattern, the accumulator will process the full command word if the central decoder address is correct. If the address is not correct, the accumulator will be cleared. Processing consists of a polynomial code check that detects all single and double bit errors and many other multiple bit errors. If the command is accepted, it is reformatted and transferred to either the stored memory for later processing, or the output section where it is encoded and transferred to a remote decoder.

An accumulator counts the number of commands accepted in each command sequence. A check bit indicates if a command has been rejected. This information is relayed to earth in the telemetry data.

Stored Time Processing. The stored command processor section (SCP) of the central decoder works as follows. The SCP memory is a long static shift register with multiple entry points. Three kinds of commands may be entered into storage: time delay, magnitude, and pulse. The commands are placed into memory after the polynomial code check and a decode that indicates the data is to be stored. The SCP is started into operation by a real time pulse command. Upon receipt of the start command, the data is shifted by a high speed clock to the most significant position of the command memory, but not until the next major spacecraft clock transition. A word is then shifted out of storage; if it is decoded to be a pulse or magnitude command, it is routed through the central decoder directly to a remote decoder. When the stored command word is a time delay, all processing from the SCP is stopped until the delay is counted out. The delay counter has 2^{16} states and counts at a 8 Hz clock rate. Words can be processed at a maximum rate of one every 125 ms. At the end of the delay time, the next word is removed from the memory and the process is repeated. This timing process allows a series of commands separated by 125 ms to be processed with only one time delay command or a series of time delay commands could be cascaded to establish a very long delay. The only prerequisite for operation is that the first word be a time delay command.

An additional feature of the SCP allows verification of the memory contents via telemetry prior to starting the SCP. The memory must be in the "stopped" mode during this readout and recycling.

Receiver Reverse Implementation. The receiver reverse function consists of a timer and logic that can provide three pulse outputs to change the connection between the antennas and receivers, and to disconnect the high gain antenna. The appropriate output is enabled upon the receipt of a pulse command or the count-out of the timer with a delay of about 32 h. The timer is reset if there is a polynomial code check verification (processing a command) or when the output is commanded into either state. The timer can be defeated by commanding the same state of the antenna/receiver connection about every 30 h from either the real time link or in the stored command processor.

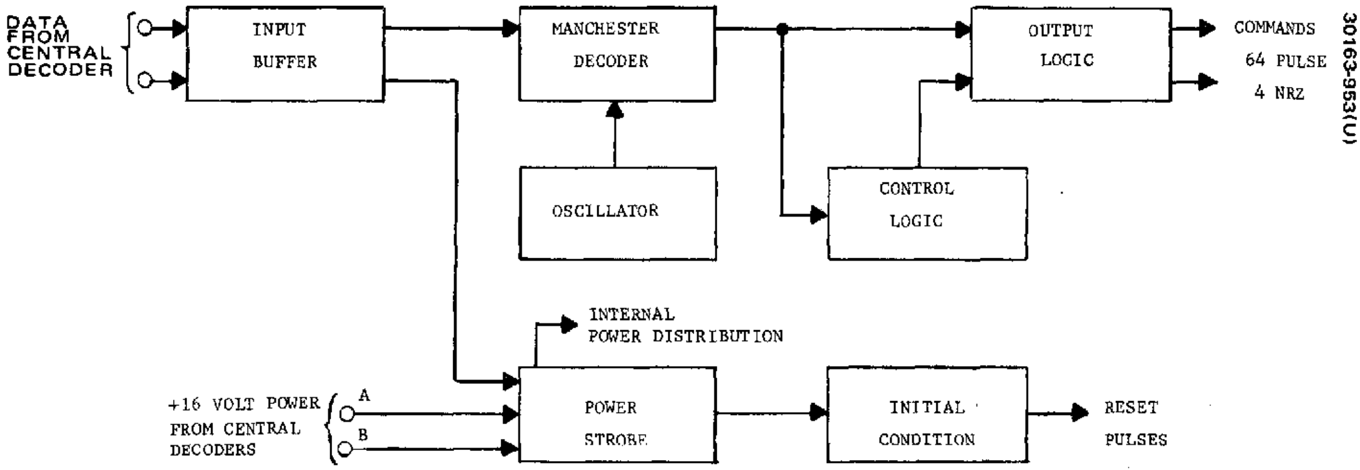


FIGURE 6-4. REMOTE DECODER BLOCK DIAGRAM

Remote Decoders

The remote decoders are from the OSO-I program and do not have to be modified for the Pioneer Venus requirements. The OSO remote decoder is designed to receive data over a single wire from either of two central decoders and distribute 4 NRZ serial magnitude signals and 64 pulse commands to users. The unit has been designed to provide adequate safeguards to prevent false commands and spurious outputs. The unit is power strobed to minimize long term power requirements.

The block diagram of the remote decoder is shown in Figure 6-4. Operation is as follows. The input buffer accepts signals from the central decoders. It provides a data line output which is high whenever the input is in the logic 1 state or when a pulse command is to be transmitted. A memory element, which is activated by the data input, controls unit power. The remote decoder is deactivated by an internal command, after the processing has been completed, or by a pulse command from another unit. The Manchester decoder converts the incoming Manchester data into a serial NRZ format. It also provides the required clock and data signals on separate output lines. The initial condition portion of the circuit resets the remote decoder logic to prevent spurious outputs. An L-C oscillator is used to generate the timing reference needed for decoding Manchester data. The control and output logic decodes the address of the output and distributes pulse and magnitude commands to the addressed output line.

Pyro Control Unit

The block diagram of pyro control unit is shown in Figure 6-5. Each driver is capable of simultaneously firing three squibs. Each driver and the associated driver in the redundant unit, can be fired simultaneously for a total simultaneous drive capability of six squibs or bridge wires. Interlocked commands are used to fire a squib; after arming the switch, the driver must be enabled within 1 sec in order to fire the squibs. When the timing interval is complete, the arming function is disarmed.

The pyro control unit will be a new packaging design, with a large percentage of the circuits being similar to those from other programs. The simultaneity requirement to fire three squibs within 2 ms required a design that would drive a minimum current of 15 A into three 1 ohm pyrotechnic loads from the same driver. Because squib firing is irreversible, two interlocked commands are used to fire a squib. Due to the high currents during firing, the squib circuits will be enabled only from the command memory; this will allow the maximum window between arming and firing commands to be set at less than a second, which will prevent any short circuits in the squibs or drivers from draining the battery for a longer time.

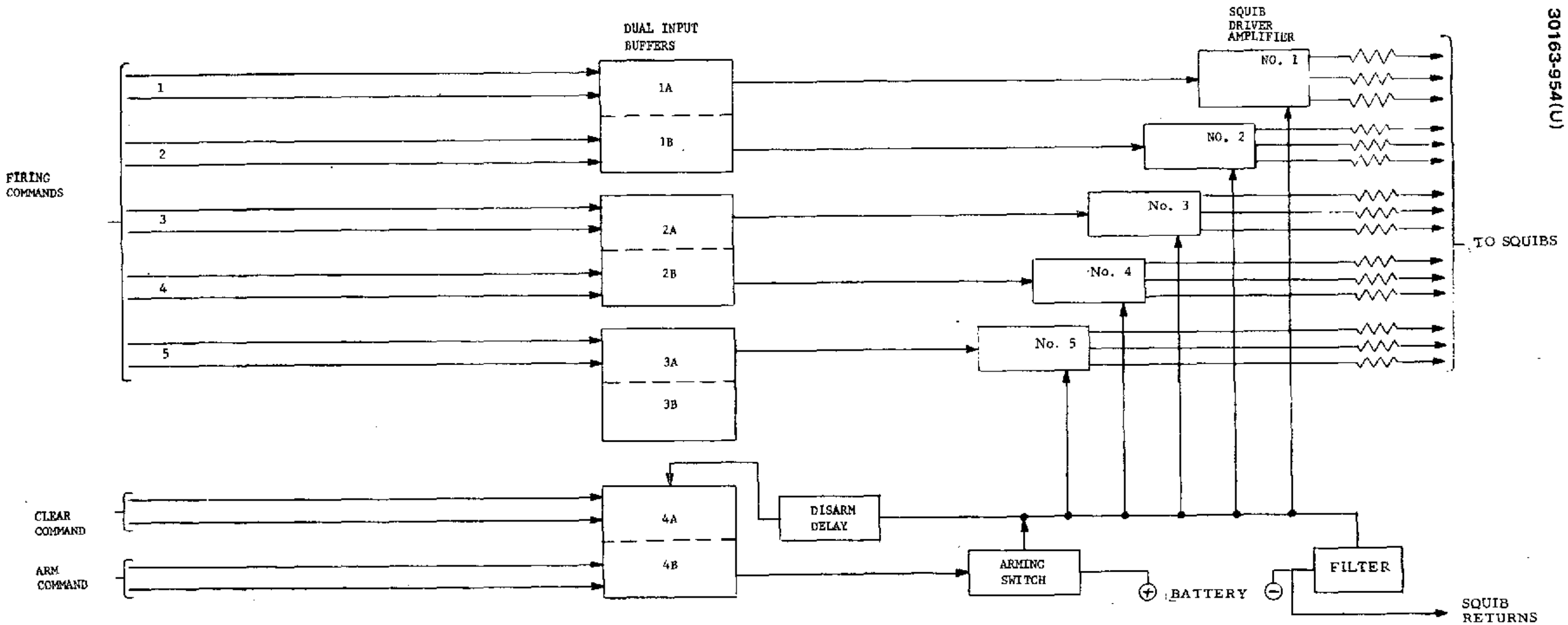


FIGURE 6-5. PYRO CONTROL UNIT

Probe Bus/Orbiter Data Handling Subsystem Functional Summary

A block diagram of the data handling subsystem is shown in Figure 6-6. This subsystem consists of four basic types of units: dual remote multiplexers, PCM encoders, telemetry processors, and data storage units that are required only on the orbiter spacecraft.

The PCM encoder and telemetry processor are redundant and fully crosstrapped to provide high reliability. Although the remote multiplexers are dual units, they are not used to provide complete dual redundancy. Redundancy is provided only for essential data inputs by crosstrapping these data sources to other multiplexer inputs and sampling the data twice within each format. The reason for using two data storage units is twofold; first, it can provide full redundancy during the mission phases in which less than half of the total data storage is required; second, it provides graceful degradation during phases of the mission that require more than half of the data storage since, if one unit fails, 50 percent of the total capacity is still usable.

The following is a brief description of the major functions performed by each of these units.

Remote Multiplexer

1) Accepts data from spacecraft subsystems and science instruments (and from probes on the probe bus)

2) Multiplexes analog, serial digital, and bilevel discrete input data onto a single output line to be transferred to the PCM encoder

3) Receives address control data from the PCM encoder to provide channel selection

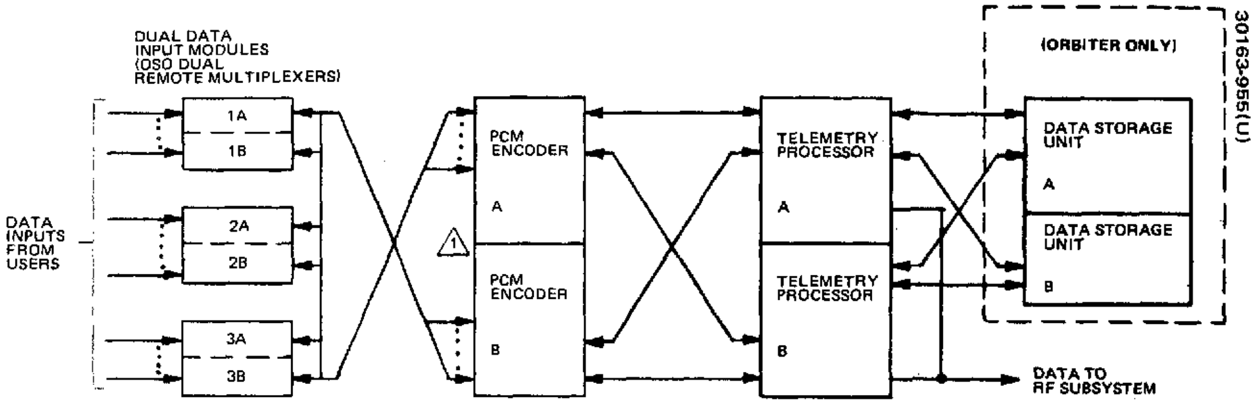
4) Provides signals to data handling subsystem users to control the sampling of digital data

PCM Encoder

1) Accepts PAM analog data and serial digital telemetry data from remote multiplexers

2) Digitizes analog data into 8 bit words and combines them with digital data into a PCM bit stream to be transferred to the telemetry processor

3) Relays address control data from the telemetry processor to the remote multiplexers for channel selection



1 INTERFACE IS FULLY CROSS - STRAPPED; EACH ENCODER COMMUNICATES WITH EACH DATA INPUT MODULE

FIGURE 6-6. MODULAR DATA HANDLING SUBSYSTEM

Telemetry Processor

1) Provides a versatile telemetry frame format which is based upon 16 preprogrammed formats which can be selected by real time or stored command.

2) Provides capability for processing simultaneously two different data streams (data formats) and at different bit rates; one data stream (real time) is directed to the RF subsystem for transmission while the other data stream (stored-time) is directed to memory for storage. (This capability is used only on the orbiter spacecraft.)

3) Generates addresses of data inputs to be sampled.

4) Encodes, modulates, and controls amplitude of PCM data and transfers it to the spacecraft transmitters or memory.

Data Storage Unit

1) Provides storage for scientific and engineering data on the orbiter spacecraft.

Table 6-14 presents a summary of the basic functional characteristics for the data handling subsystem. It shows the similarity and differences between the probe bus and orbiter spacecraft. As indicated in the table, the characteristics are identical with four exceptions. The major difference is that data storage is only required on the orbiter spacecraft; the other three differences concern the data formats. Only the telemetry processor is affected by these functional differences. In spite of these differences, the telemetry processor hardware is the same on the two spacecraft.

Probe Bus/Orbiter Data Handling Subsystem Hardware Design Summary

The derivation and characteristics of the data handling subsystem hardware are shown in Table 6-15. The table briefly describes the source and type of hardware employed, in addition to presenting the mass, power, and volume characteristics for both the probe bus and orbiter spacecraft.

Three of the data handling units are planned to be built by Hughes. Of these units, the remote multiplexer and PCM encoder units are existing OSO units, whereas the telemetry processor is a new design that employs OSO circuits to a large degree. The remote multiplexer and telemetry processor employ a large amount of existing custom LSI circuitry as a means to reduce mass.

The data storage unit is planned to be purchased from Electronic Memories. A previously designed core memory unit of the company's SEMS will be modified to meet the storage and interface requirements.

TABLE 6-14. PROBE BUS/ORBITER SPACECRAFT DATA HANDLING SUBSYSTEM FUNCTIONAL CHARACTERISTICS SUMMARY

Parameter	Characteristics: Probe Bus	Characteristics: Orbiter
Word size, bits	8	
Number of data inputs	192	
A/D conversion accuracy	±0.4 percent (8 bits)	
Data storage capacity, megabits	None	1.048
Data bit rates, bps	8 to 2048	
Data formats		
Subframe length	32 words	
Telemetry frame length	512 words, 16 subframes	
Number of preprogrammed (ROM) 32 word subframes	16 bus unique	16 orbiter unique
Number of downlink formats	16 bus unique	16 orbiter unique
Number of data storage formats	None	16
Error code	Convolutional, length 32, rate 1/2	
Modulation type	PCM/PSK	
Subcarrier frequency, Hz	(TBD)	(TBD)

TABLE 6-15. ATLAS/CENTAUR MODULAR DATA HANDLING SUBSYSTEM HARDWARE DERIVATION AND CHARACTERISTICS

Unit (Quantity)	Derivation		Characteristics*				
			Mass		Power, W	Volume	
	Probe Bus	Orbiter	kg	lb		cm^3	in^3
Data input module (3) (remote multiplexer)	OSO (Three dual units; existing LSI; PC boards)	OSO (Three dual units; existing LSI; PC boards)	1.1	2.4	0.4	900	55
PCM encoder (2)	OSO (Two single units; existing MSI; MICAM,** PC boards, and modules)	OSO (Two single units; existing MSI; MICAM,** PC boards, and modules)	3.1	6.8	2.6	3,510	214
Telemetry processor (2)	New design (Uses 70 percent OSO circuits; no new LSI; MICAM and modules)	New design (Uses 70 percent OSO circuits; no new LSI; MICAM and modules)	3.6	8.0	5.7	6,770	413
Data storage unit (2)	Not required	Core memory-POV (Modified electronic memories design)	– 9.1	– 20.0	– 3.0	– 9,630	– 588
Probe bus totals			7.8	17.2	8.7	11,180	682
Orbiter totals			16.9	37.2	11.7	20,810	1,270

*Where two sets of values are given, the bottom line is for the orbiter spacecraft.

**MICAM is the acronym for Hughes "microelectronic assembly method."

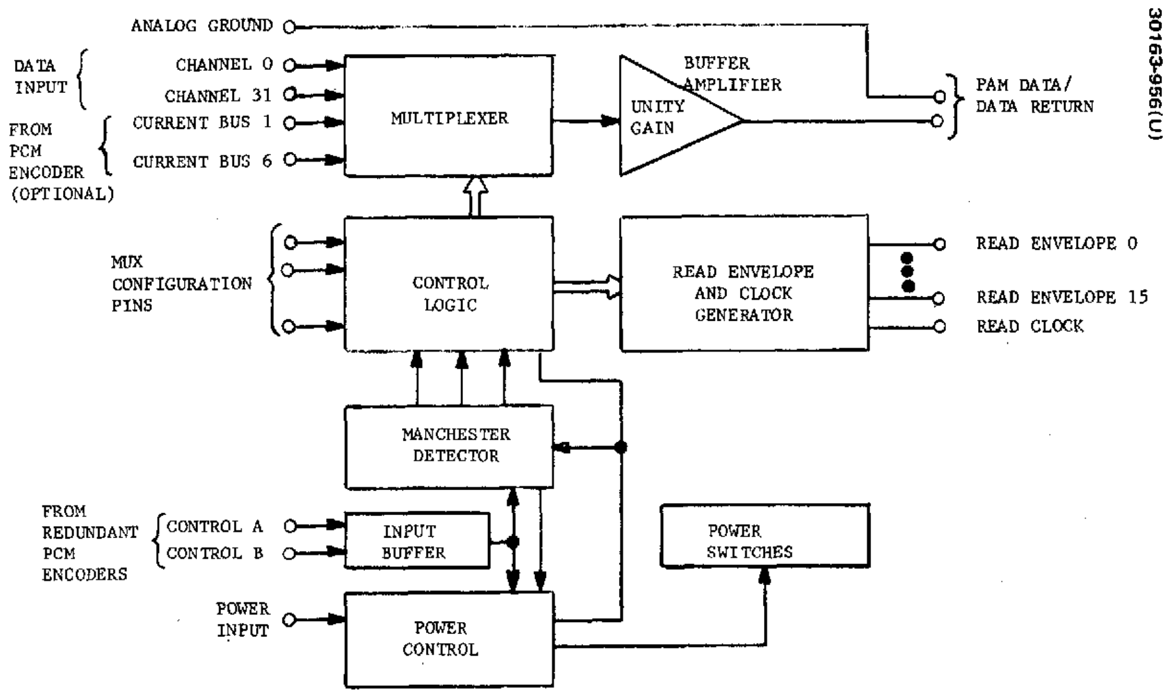


FIGURE 6-7. REMOTE MULTIPLEXER MODULE

Probe Bus/Orbiter Data Handling Subsystem Description

Remote Multiplexers

The remote multiplexer module is an existing design from the OSO-I program; it does not require any modification.

A block diagram of this data input module is shown in Figure 6-7. It is basically a 32 input random access commutator with a single (but redundant) input control line and PAM output data line having an associated data return. The unit is capable of accepting high level analog data, parallel bilevel digital data, and serial digital data in various ratios. Upon request from the PCM encoder, analog data is multiplexed onto the PAM output line. Parallel bilevel data are sampled in 8 bit bytes and are gated to the output data line as an 8 bit serial word. Serial digital data is transferred to the unit in 8 bit bytes and is gated to the output data line through use of a gated read clock and read envelope signals which are provided by the remote multiplexer module. The input control line is used for both power on/off control and channel address information. The serial address into the unit is Manchester coded such that a coherent shift clock can be derived from the serial address, resulting in a single wire control. Sixteen read envelope signals are generated in synchronism with the sampling of each 8 bit serial or parallel digital word, thus informing the user that his data is being sampled. Serial data users must logically "AND" the read clock output with the appropriate read envelope signal to enable data readout.

The multiplexer output is connected to a unity gain voltage follower amplifier for the purpose of presenting a high input impedance to the multiplexer data sources while providing a low output impedance; thus, the noise susceptibility of the output PAM data line is minimized. The PCM encoder treats all data from a given remote multiplexer differentially with respect to each multiplexer reference ground, thus eliminating ground offsets between the data source and the PCM encoder's analog-to-digital converter.

The multiplexer also contains six current switches that operate synchronously with the voltage switching gates. The purpose of the current switches is to commutate a precision constant current source (1.000 mA) into the selected multiplexer channel; this feature thus provides signal conditioning of temperature sensors (such as thermistors or platinum resistors), pressure or position sensitive potentiometers, and other transducers directly. The current source is provided by the PCM encoder if this conditioning is required.

The multiplexer configuration pins select the mode (i.e., the channel assignments for the three types of input data) in which a particular multiplexer is to be operated.

PCM Encoder

The PCM encoder is an existing design from the OSO-I program; it also does not require any modification.

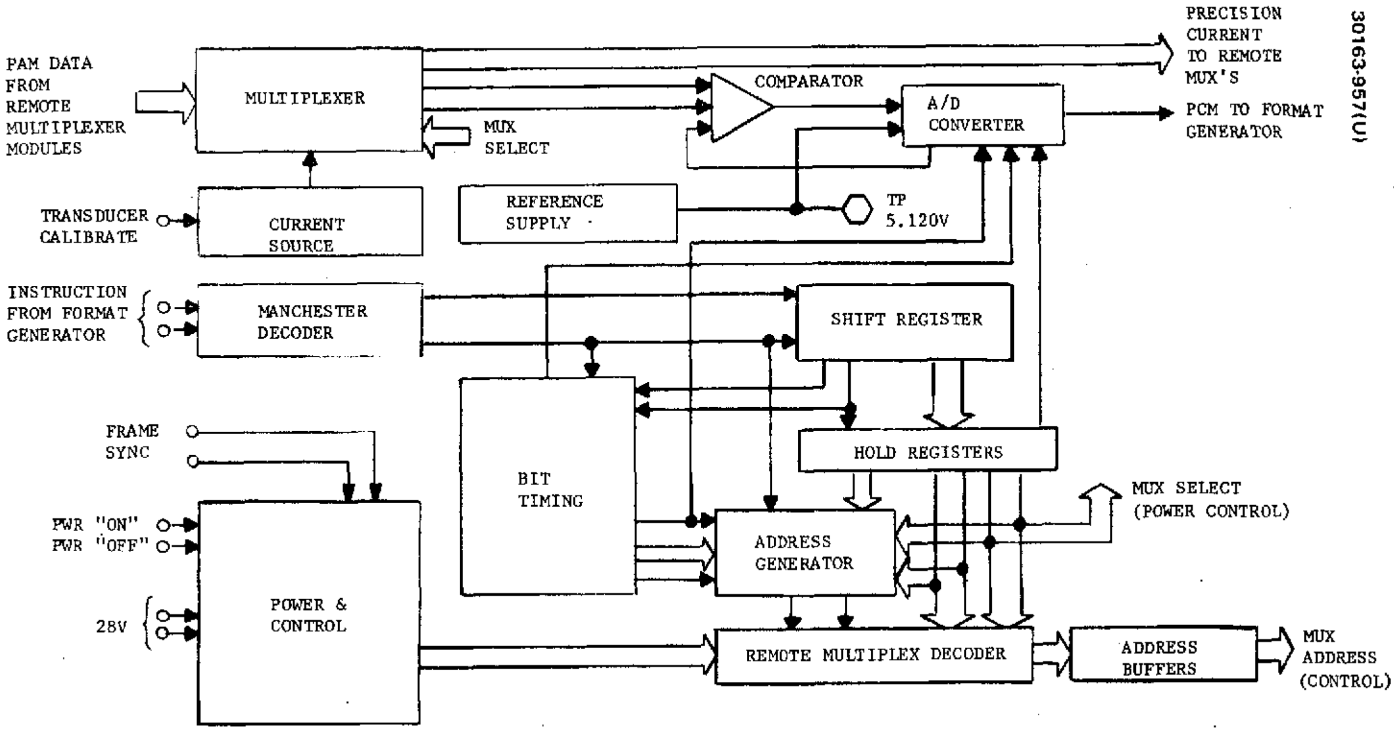


FIGURE 6-8. PCM ENCODER

A block diagram of the PCM encoder is shown in Figure 6-8. The unit accepts PAM analog and serial digital telemetry data from the remote multiplexers, processes and synchronizes the data, and generates a non-return to zero pulse code modulated (NRZ-L/PCM) output bit stream to the telemetry processor. Analog-to-digital conversion of the remote multiplexer analog data is also provided. The PCM encoder relays the necessary address control data from the telemetry processors to the remote multiplexers for channel selection. The PCM encoder also provides regulated power and on/off commanding of the remote multiplexers. Primary timing and control of the PCM encoder is derived from the Manchester coded (split phase) instruction address provided by the telemetry processors.

The remote multiplexer decoder circuit decodes the remote select bits and gates the output and power control pulses onto the appropriate output control lines whenever remote multiplexer power control commands are received.

The multiplexer differentially multiplexes the PAM data and return from the remotes into the A/D comparator. It also steers a precision 1.000 mA constant current source to individual output lines. A calibration resistor, which is wired to the unit connector for transducer calibration, can be wired to a remote multiplexer telemetry input for calibration of the current source.

The analog-to-digital converter accepts the PAM input data from the remote multiplexers and encodes it into a serial 8 bit digital equivalent of the high level analog data (MSB outputted first) or behaves as a fixed threshold detector for making one-zero decisions on the incoming digital data. The treatment of the data (as analog or digital) is controlled by the A/D bit in the input control word.

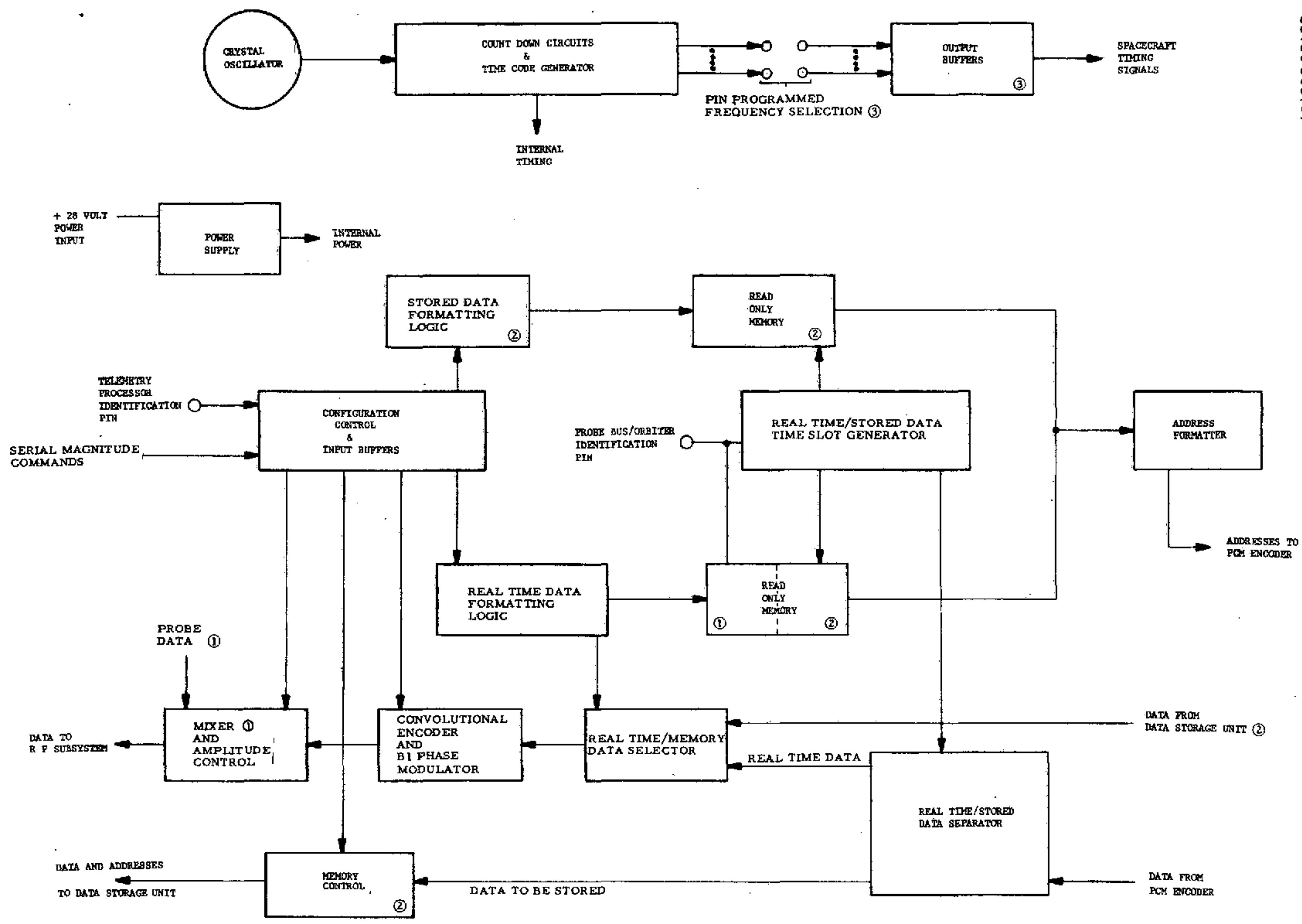
If digital data is to be processed, the comparator is forced to a constant 2.56 V reference. Thus, the comparator registers a logic 1 for data above this voltage and a logical 0 for data below this voltage.

All power supplies for the PCM encoder are generated internally from the spacecraft 28 V bus. A series regulator provides regulation of the bus voltage and an electronic conversion unit (ECU) generates all of the supply voltages required by the unit, including power to be distributed to the remote multiplexers.

Telemetry Processor

A block diagram of the telemetry processor unit is shown in Figure 6-9. The unit is a new packaging design but uses a large number of existing OSO circuits. It provides three basic functions: spacecraft clock and timing signals, real time processing and stored format processing.

Operation of the telemetry processor is briefly described below.



LEGEND OF FUNCTIONAL DIFFERENCES BETWEEN PROBE BUS AND ORBITER:

① FUNCTIONS/HARDWARE USED ONLY ON PROBE BUS
② FUNCTIONS/HARDWARE USED ONLY ON ORBITER
③ FUNCTIONS/HARDWARE USED UNIQUELY ON BOTH SPACECRAFT

NOTE: UNIT IS IDENTICAL ON BOTH SPACECRAFT EXCEPT FOR PIN PROGRAMMING AS NOTED

FIGURE 6-9. TELEMETRY PROCESSOR

Spacecraft Clock. The oscillator and countdown circuits provide timing for the spacecraft subsystems and science instruments. In addition to various clock signals, a time code generator accumulates time and transfers the data into the telemetry stream upon request from the format generator.

Real Time Processing. Real time processing is accomplished by selection of a format via the command subsystem and configuration controller. The same path is used to select the bit rate, output voltage level, and convolution encoder bypass operation. The format and address information are in the read only memory and, upon selection, address information is fed to the address formatter and relayed to the PCM encoder. Data returned from the PCM encoder is routed to the convolution encoder and biphase modulator. It is then amplitude controlled and mixed with data from the probes (probe bus only) and sent to the RF subsystem.

Stored Format Processing. Stored format processing is used only on the orbiter spacecraft. It is accomplished by using a parallel path with real time processing, except there are separate time slots provided in order to time share the same remote multiplexers. Stored time data received from the PCM encoder is separated from real time data and fed to the data storage unit via the memory control.

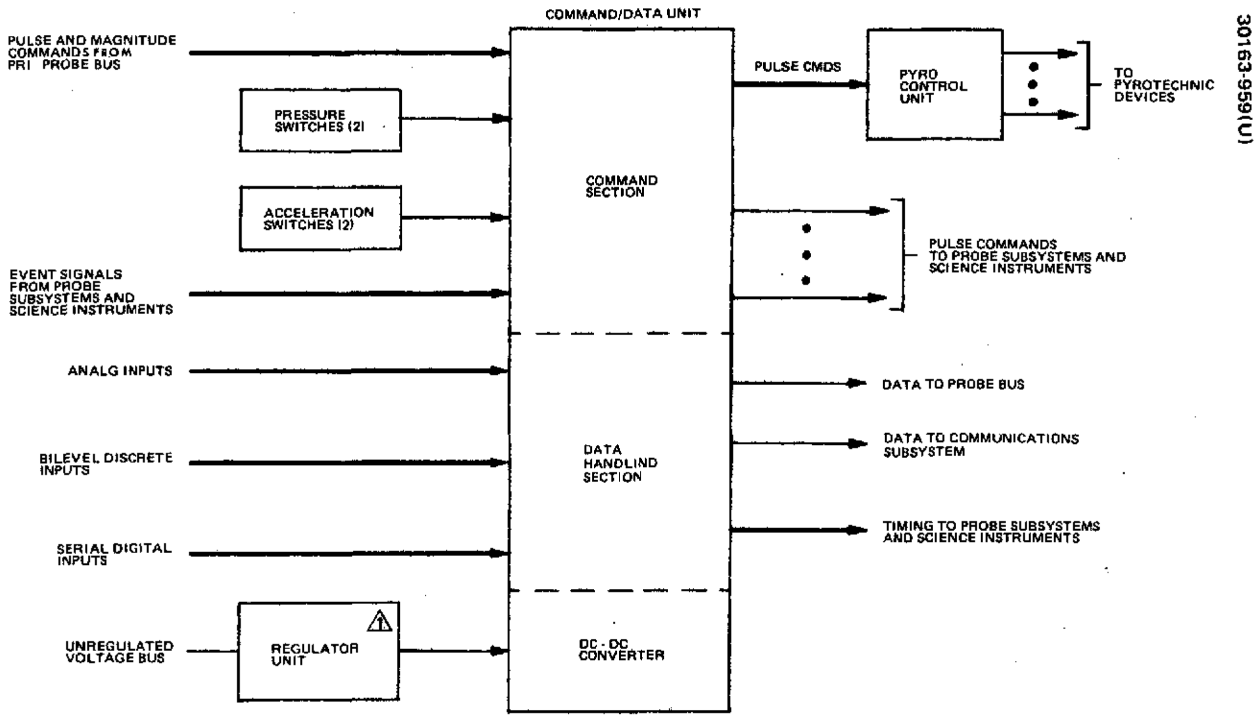
Data Storage Unit

The orbiter spacecraft data storage unit will be a purchased item. The baseline is a magnetic core memory. The memory will be constructed in two equal sections, each with half of the required mission capacity. Because the total memory is not redundant, failure of one half of the memory will allow graceful degradation by allowing much of the data to be recovered.

6.4 PROBE COMMAND AND DATA HANDLING SUBSYSTEMS

The Atlas/Centaur command and data handling subsystem for the probes are functionally quite similar to those previously described for Thor/Delta in Section 5. However, there are a number of significant hardware design differences, the primary one being that custom LSI circuit technology is not employed in this Atlas/Centaur version of the subsystems. The major hardware and functional differences were previously described in section 6 and subsystem 6.2. Differences in requirements were described in subsection 6.1; the principal differences are in the magnitude of certain requirements such as number of data inputs, sample rates, bit rates, data storage bits, number of commands, and pyrotechnic initiators fired.

As with the Thor/Delta design, the command and data handling functions are combined into a single subsystem in both the large and small probes. The subsystems are functionally quite similar on the two vehicles; the differences between the probes are explained in the following discussion.



⚠ REGULATOR UNIT IS SEPARATE ON SMALL PROBE; THE REGULATOR IS CONTAINED WITHIN THE COMMAND/DATA UNIT ON LARGE PROBE.

FIGURE 6-10. LARGE AND SMALL PROBES COMMAND AND DATA HANDLING SUBSYSTEM

Command and Data Handling Subsystem Functional Summary

Figure 6-10 is a functional block diagram of the command and data handling subsystems; it represents the subsystems for both the large and small probes, although their physical designs are different. The subsystems are divided into two control items or units: the command data unit and the pyro control unit. The following is a brief description of the major functions performed by these subsystems.

The command portion of the subsystem does the following:

1) Accepts pulse and magnitude commands from the probe bus spacecraft for preseparation tests and for initialization of the probe's cruise timer

2) Provides an accurate timer to initiate probe preentry sequences after a 20 to 23 day cruise period

3) Provides pulse command outputs to probe subsystems and scientific instruments based upon predetermined (stored) sequences and real time events

4) Detects pressure and acceleration events during descent to initiate certain command sequences

5) Provides high current switching capability for commanded firing of pyrotechnic devices.

The data handling portion of the subsystem does the following:

1) Accepts analog, bilevel discrete, and serial digital data from the probe subsystems and science instruments

2) Digitizes analog data into 10 bit serial words

3) Formats all data according to one of several stored formats

4) Convolutionally encodes and biphase modulates the PCM data

5) Provides for data storage during entry blackout.

Table 6-16 summarizes the basic functional characteristics of the command and data handling portions of the subsystems for the large and small probes. As indicated in the table, many of the characteristics are identical for both probes. The differences that exist are primarily due to the fact that the small probe is essentially a subset of the large probe; the small probe requires less command and data handling activities to satisfy its mission requirements.

TABLE 6-16. LARGE/SMALL PROBE COMMAND AND DATA HANDLING SUBSYSTEMS FUNCTIONAL CHARACTERISTICS SUMMARY

Parameter	Characteristics: Large Probe	Characteristics: Small Probe
Command		
Cruise timer		
Range	24 days	
Stability (0° to 40°C)	1×10^{-5} (20 sec in 23 days)	
Resolution	2 sec	
Descent timer		
Range	13.6 min (between recycle)	
Stability (-20° to +65°C)	5×10^{-5}	
Resolution	0.1 sec	
Command mode	Stored	
Command type	Pulse	
Command initiation	Time or event	
Number of event inputs	8	
Number of command outputs	48	32
Total command executions	256	
Number of pyrotechnic drivers	4 redundant pairs, multiple 2 redundant pairs, single 10 nonredundant, single	3 redundant pairs, multiple
Maximum delay between concurrent fire pulses	0.1 ms	
Number of pyro initiators fired (capability)	38	18
Number of acceleration detection switches	2	2
Number of pressure detection switches	2	2
Data handling		
Word size, bits	10	
Number of analog inputs		
10 bit accuracy	16	8
8 bit accuracy/10 bit resolution	32	24
Number of digital inputs		
10 bit serial	16	8
Bilevel discrete	32	24
Data storage capacity, bits	2048	512
Number of data formats	3	
Data bit rates, bps	160 or 80 20 (store only)	60 or 30 or 10
Error coding	Convolutional, length 32, rate 1/2	
Modulation type	PCM/PSK	
Subcarrier frequency, Hz	20,480	30,720

Command and Data Handling Subsystem Hardware Design Summary

A summary of the hardware derivation and characteristics for the command and data handling subsystems is shown in Table 6-17. In addition to presenting the mass, power, and volume characteristics for the units in both the large and small probes, the table briefly describes the source and type of hardware employed. Wherever practicable, the units make use of presently qualified components, utilize existing designs, and employ common designs between the large and small probe subsystems; these features all contribute to a lower cost design.

New design will be undertaken for the command/data units on both probes. There are two principal reasons for not using existing command and data handling hardware. First, certain performance requirements, such as the 10 bit analog-to-digital conversion accuracy required, and certain environmental constraints, such as the high deceleration experienced upon entering the Venus atmosphere, are not readily met by existing equipment, thus requiring some cost to modify the equipment and possibly some redesign. The second reason is that the advantage of using a custom design in the relatively small probe vehicles, from a mass and volume standpoint, becomes apparent when one considers that any inefficiency is multiplied by a factor of four, the total number of probes. The latter consideration is also important in the decision to make the small probe design customized rather than forcing the larger capability of the large probe command/data unit upon each of the three small probes. It is important to note, however, that a great deal of commonality is employed between the large and small probe designs in order to minimize design costs.

The pyro control units contain squib drivers of a new design, each of which is capable of simultaneously firing three squibs. Because of the large number of pyrotechnic initiators on the probes, the use of the multiple squib driver results in a considerable mass and volume reduction. Cost is minimized due to the fact that the designs used for the probe bus/orbiter spacecraft and the large and small probes are nearly identical, the principal difference being in the packaging design.

Command/Data Unit Description

The command/date unit performs all command and data handling functions with the exception of firing pyrotechnic devices, which is done by the PCU. Figure 6-11 is a detailed block diagram of the command/data unit and is applicable to both the large and small probes. Maximum commonality of design is used between the two probes. Differences in hardware exist in the number of multiplexer elements, the number of memory elements used for data storage, format programming, and command sequencing circuitry, and in the number of command decoding and output buffering elements. The relative magnitude of the difference in hardware can be seen by reference to Tables 6-16 and 6-17.

The command/data unit can be functionally divided into the command section and the data handling section, each of which is described below.

TABLE 6-17. ATLAS/CENTAUR PROBE COMMAND AND DATA HANDLING SUBSYSTEM HARDWARE DERIVATION AND CHARACTERISTICS

Unit	Derivation	Characteristics: Mass, kg	Mass, lb	Power Cruise/Descent, W	Volume, cm^3	Volume, in^3
Large probe						
Command/data unit	New design, existing technology; MICAM,* PC boards, and modules	1.95	4.3	0.04/8.3	2720	166
Pyro control unit	New design, existing technology; 50 percent probe bus circuits; modules	1.18	2.6	(transient only)	1330	81
Acceleration switches (2)	(1) Existing design; space qualified (1) Existing design; must be space qualified	0.23	0.5	–	66	4.0
Pressure switches (2)	Existing design, high reliability; must be space qualified	0.18	0.4	–	25	1.5
Total		3.54	7.8	0.04/8.3	4140	253
Small probe						
Command/data unit	New circuit design; 70 percent large probe circuits New product design; MICAM,* PC boards and modules	1.54	3.4	0.04/4.6	2280	139
Regulator unit	New package; existing circuit design; modules	0.18	0.4	0.0/1.2	197	12
Pyro control unit	New circuit design; 80 percent large probe circuits New product design; modules	0.55	1.2	(transient only)	361	22
Acceleration switches (2)	(1) Same as large probe; space qualified (1) Existing design; must be space qualified	0.23	0.5	–	66	4.0
Pressure switches (2)	(1) Same as large probe (1) Existing design, high reliability; must be space qualified	0.18	0.4	–	25	1.5
Total		2.68	5.9	0.04/5.8	2930	179

*MICAM is the acronym for Hughes "microelectronic assembly method."

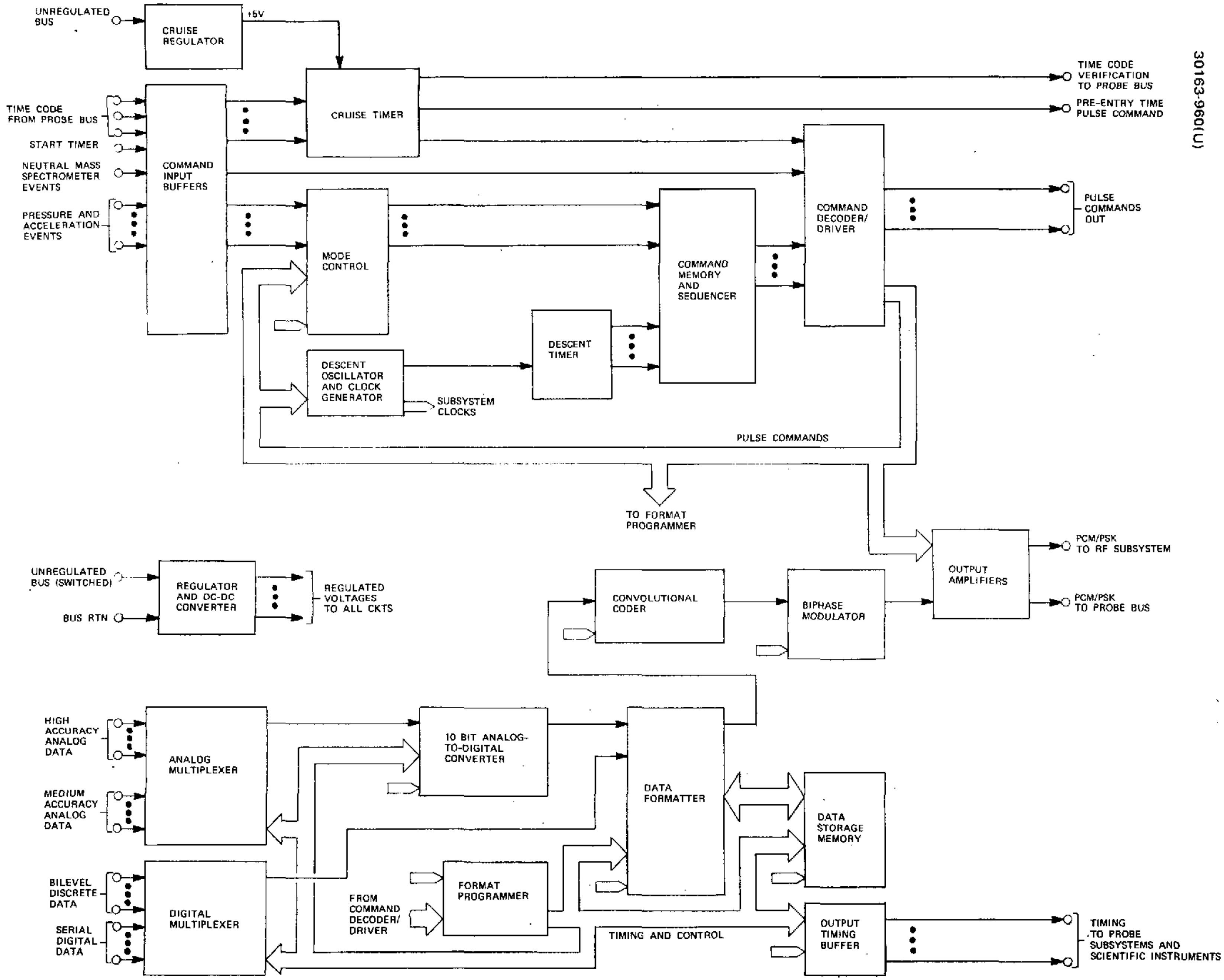


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FIGURE 6-11. LARGE AND SMALL PROBES COMMAND/DATA UNIT

Command Section

The command section of the probe command and data handling subsystem receives pulse and magnitude commands from the probe bus prior to separation of the two vehicles. These commands are used to start or stop preseparation tests, to provide a 20 bit time code to the probe's cruise timer, and to start the timer which, in turn, initiates preentry events with an accuracy of ±20 sec following a nominal 20 day cruise period. After separation of the probes from the probe bus, all pulse commands generated by the command section result from fixed stored commands or from "event" occurrences signaled by probe subsystems and science instruments. No magnitude commands are provided by the probe command section.

The command memory has the capacity to store 64 output commands and 256 individual command executions which are programmed into a read only memory prior to unit assembly. Stored commands are activated by either time or events. The delays are programmed by time tags stored as part of the command word in memory. These time tags are referenced to the probe's descent timer for execution timing control. All time tagged commands will be executed relative to special event occurrences such as acceleration or pressure switch closures. When an event occurs or when time coincidence occurs between the stored command time tag and the binary state of the descent timer, stored commands are processed through the command decoder. One of two timer resolution modes is commanded internally. In the high resolution mode, command execution can be provided with a resolution of less than 100 ms. In the low resolution mode, which is used for most of the descent phase of the probe mission, command execution is provided with a resolution of less then 800 ms.

One of the large probe science instruments, the neutral mass spectrometer (NMS), requires squib firing commands which are independent of probe acceleration sensing or atmospheric pressure sensing. Since all stored command executions during entry and descent are relative to these two parameters, NMS valve ON/OFF commands will be generated independent of the stored command sequencer; they are initiated upon receipt of serial pulses from the NMS instrument.

Standard pulse command outputs from the decoder are used to activate squib driver circuits for firing of the pyrotechnic squibs associated with probe operations and with the science instruments. A two command sequence is required for squib firing; the first command applies unregulated power to arm all squib drivers in the pyrotechnic control unit; the second is a unique command pulse to fire individual squib drivers.

The primary interface between the command section and the using subsystems and science instruments is at the output of the command decoder. The decoder is deenergized during the 20 day cruise period, except during postseparation test (approximately 10 min) and during a brief interval 2 days prior to entry. It is turned ON by the application of power to the command section, which is controlled by the cruise timer after separation. The decoder is energized for the entire descent phase beginning 15 min prior to entry.

Only one command output is independent of the command decoder: the "Engineering-ON" command normally generated by the cruise timer and which results in activating the power switches in the probe's power subsystem. All command section circuits, with the exception of the cruise timer and regulator, are powered from the switched, unregulated bus and thus are deenergized until the Engineering-ON command is executed.

An additional interface between the command subsystem and the using subsystems and payload instruments is via event signals. These signals originate as pulse outputs from the using subsystems and instruments and are sent to the command section to initiate time sequenced commands. Time sequenced commands are executed at preprogrammed times relative to the occurrence of these event signals. The command section has provisions for eight different event signals and associated event initiated command sequences. Provision is made to accept ten serial pulses on a single input and to provide asynchronous firing of 10 NMS valve squibs independently of the command memory and descent timer.

Pulse command outputs are grouped within the command section in multiples of 16 signals. Each group is referenced to its own isolated signal return, which can be made available to users who utilize all 16 pulse commands of a single group. A group divided among many users will have its return tied to the command section signal ground. In either case, the transmission of a logical "1" relies upon the eventual commonality of the command section signal ground and the user's signal ground (i.e., a return path for the pulse command current via a system or "unipoint" ground). Noise immunity provided by the standard input buffer (discussed in the command subsystem interface definition of subsection 4.1) is sufficient so that providing command signal returns to users is unusually unnecessary. These returns are not provided unless abnormally high voltage differentials exist between the command section signal ground and the user's ground.

Data Handling Section

The data handling section of the command and data handling subsystem accepts analog, serial digital, and bilevel discrete data from probe subsystems and science instruments. It digitizes the analog data and formats all data into a serial bit stream. Formatting is accomplished by use of ROM elements; multiple formats are provided to improve the efficiency of downlink data transmission. Accelerometer data is stored during the high deceleration experienced upon initial encounter with the Venus atmosphere; it is later played back and formatted with real time data to be transmitted to earth. The data handling section convolutionally encodes the serial data bit stream and biphase modulates it before transferring it to the communications

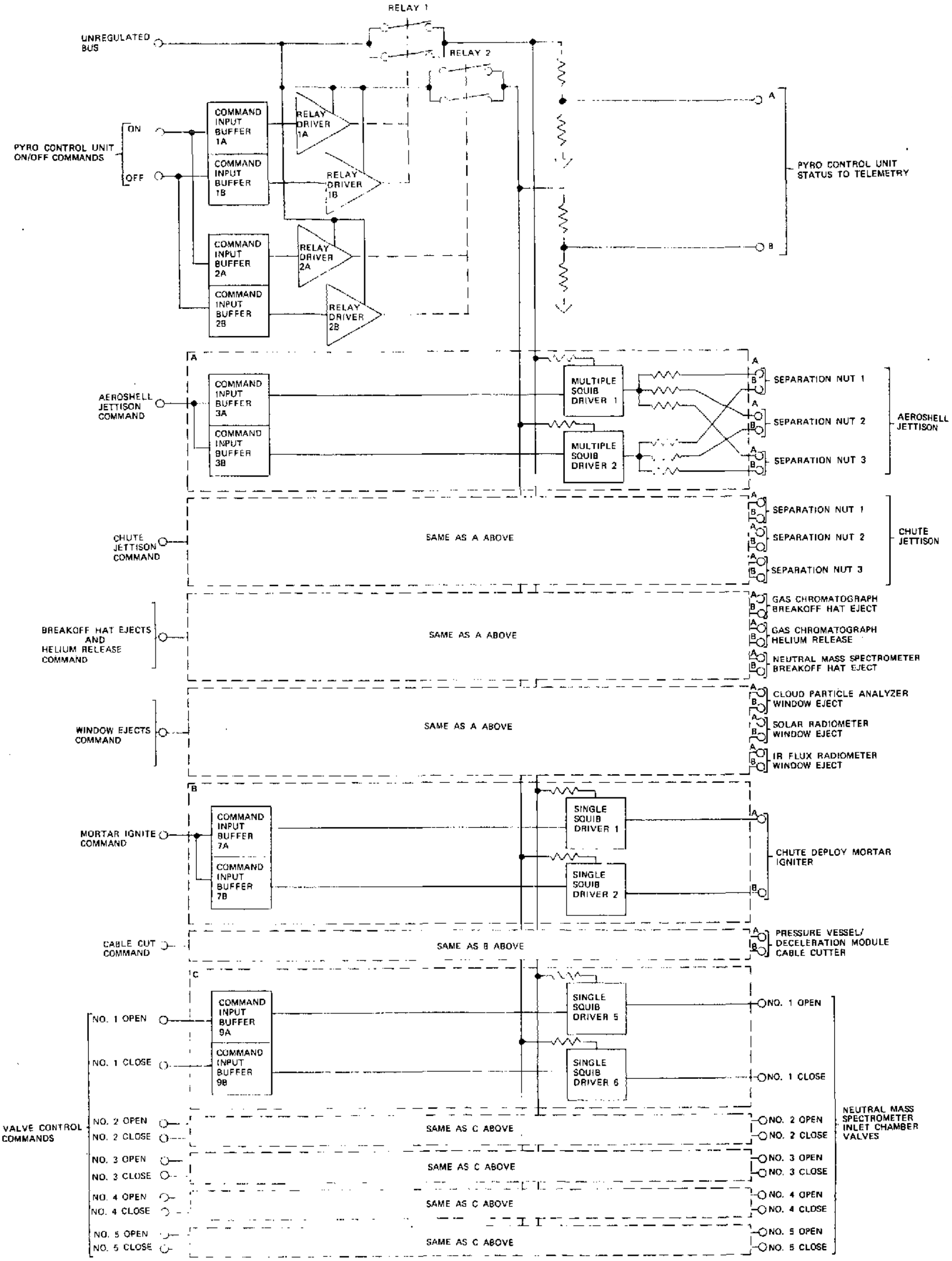

FIGURE 6-12. LARGE PROBE PYRO CONTROL UNIT

subsystem (and the probe bus prior to separation). Timing signals are provided to probe subsystems and instruments in order to synchronize transfer of data or other operations with the data handling section.

The data handling multiplexer has 96 input channels in the large probe and 64 input channels in the small probe. The distribution of analog, bilevel digital, and aerial digital channels was given in Table 6-6. The multiplexer provides special circuitry for certain analog inputs requiring high accuracy analog-to-digital (A/D) conversion. The A/D converter in the data handling section converts the sampled analog data into a 10 bit serial digital word format. Within the full scale A/D conversion range, specified input signals will be converted with an accuracy of ±5 mV or ±0.1 percent of full scale (±2.5 mV offset and ±2.5 mV quantization error). All other analog inputs will be converted with 10 bit resolution, but with ±20 mV or ±0.4 percent of full scale accuracy (±17.5 mV offset and ±2.5 mV quantization error).

PCU Description

The PCU accepts pulse commands from the command/data unit and provides high current solid state switches to switch unregulated bus current to probe subsystem and instrument pyrotechnic devices. Bus voltage to the current drivers is switched ON and OFF by relays. The relays serve a two-fold purpose: they function as "ARM" switches to ensure safe and reliable pyrotechnic firing and they block leakage current to the PCU during the probe 20 day cruise so as to eliminate any drain on the battery other than that required for the cruise timer.

All pyrotechnic operations, with the sole exception of the neutral mass spectrometer (NMS) ON/OFF valves, have redundant arming switches and firing switches in the PCU. The NMS has ten pyrotechnic valves which will not be provided with redundant arming or firing.

Large Probe PCU

The large probe PCU provides firing pulses to 38 pyrotechnic initiators. Twenty-two drivers are included in the PCU; each of eight of these drivers has the capability of simultaneously firing three squibs; and 14 drivers have single squib firing capability. Figure 6-12 is a block diagram of the large probe PCU.

Small Probe PCU

The small probe PCU provides firing pulses to 14 pyrotechnic initiators. Six drivers are included in the PCU; they each have the capability of firing three squibs simultaneously, although four of them are presently only required to fire two squibs simultaneously. Figure 6-13 is a block diagram of the small probe PCU.

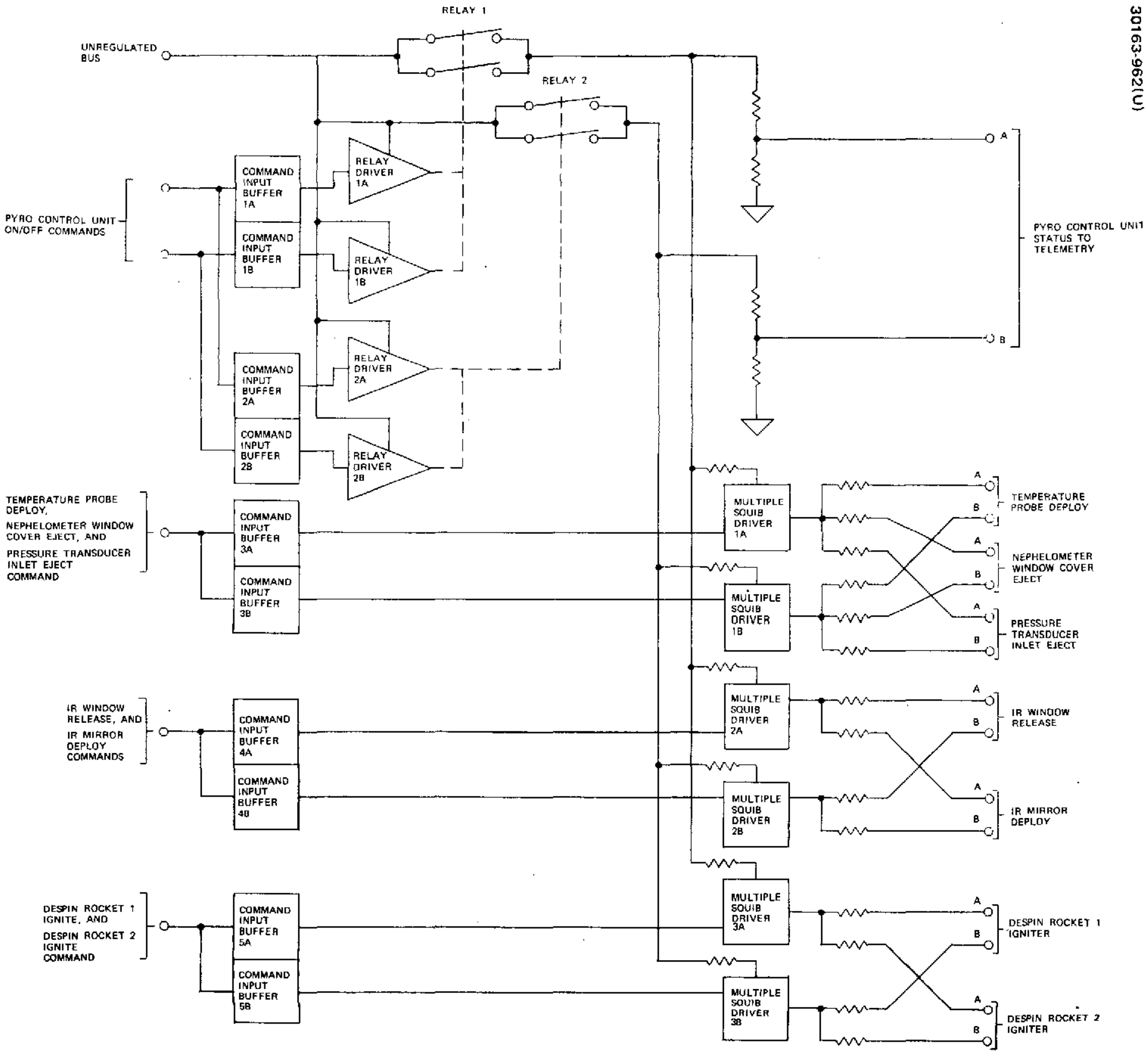

FIGURE 6-13. SMALL PROBE PYRO CONTROL UNIT

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